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General-Utility Spacecraft and Multiple-Orbit/Payload Launch Applications in Space Research and Development

JULY 1967

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Prepared for SPACE AND MISSILE SYSTEMS ORGANIZATION
AIR FORCE SYSTEMS COMMAND
LOS ANGELES AIR FORCE STATION
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GENERAL-UTILITY SPACECRAFT AND MULTIPLE-ORBIT/PAYLOAD LAUNCH APPLICATIONS IN SPACE RESEARCH AND DEVELOPMENT

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July 1967

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FOREWORD

This report is published by the Aerospace Corporation, El Segundo, California, under Air Force Contract No. F 04695-67-C-0158 and documents research carried out from September 1965 through June 1967. On 31 July this report was submitted to Lt W. E. O'Brien, SMTRE, for review and approval.

Approved

M T. Weiss

M. T. Weiss, Acting Director
Program Support

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

William E. O'Brien

W. E. O'Brien, Lt, USAF
Project Officer

ABSTRACT

Costs for the majority of near-earth, unmanned, space research and advanced development missions of the late 1960's and early 1970's can be significantly reduced by using multiple-orbit/payload launches involving general-utility spacecraft and orbital buses. This concept has evolved through the implementation of the new DOD Space Experiments and Flight Support Program (SEFSP). The modification and combination of previously developed spacecraft with other off-the-shelf space flight proven hardware to synthesize in "tinker toy" fashion a general-utility spacecraft family for use in R&D programs of this nature is discussed. The current characteristics and growth potential of the low cost, general-utility OV spacecraft family (OV1, 2, 3, and 5) which utilize off-the-shelf hardware to a maximum extent are described. The concept of the orbital bus is developed. A typical R&D program involving four spacecraft, each from a different agency, is used to show that total overall program cost can be reduced by as much as 55% through the use of multi-agency, multiple-orbit/payload, single launch vehicle missions involving orbital buses. Hypothetical, typical multiple-orbit/payload missions on both large and small launch vehicles are described.

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I. Introduction

Several unique spacecraft concepts have evolved as a result of the implementation of the new DOD Space Experiments and Flight Support Program (SEFSP). These are: the "general-utility" spacecraft family, the orbital bus, and the multiple-orbit/payload launch mission. It is the purpose of this paper to discuss the salient features of these concepts, which have not yet been fully exploited. The discussion will be restricted to the application of these concepts to research and advanced development type payloads in near-earth ($\leq$100 k n mi) orbits for missions of the late 1960's and early 1970's. Their applications to other missions such as manned, lunar, planetary, and recoverable missions or to operational communication, meteorological, and geodetic missions has not been investigated and are not considered.

The SEFSP was formed in 1967 by consolidating the Aerospace Research Support Program (ARSP) and the Space Experiments Support Program (SESP). The ARSP is managed by the AF Office of Aerospace Research (OAR), while the SESP is managed by the AF Space and Missile Systems Organization (SAMSO). The objectives of the SEFSP are to evaluate, order, integrate, and fly selected DOD tri-service and NASA aerospace experiments ranging from fundamental space physics research to certain operational DOD payloads. Because of this diversity of experiments and tests, the SEFSP deals with an unusual conglomeration of unrelated and annually changing payloads. From a systems engineering viewpoint, this continuous flux of experiments presents an unusual challenge: integrate X number of payloads on Y number of spacecraft and Z number of launch vehicles in a cost-effective and timely manner. It is this challenge that has stimulated the development of the concepts highlighted herein.

II. General-Utility Spacecraft Family Concept

A. History

In early 1965, studies were initiated under the SESP to evaluate the concept of a spacecraft design which would be adaptable on a short lead time to a variety of payloads, launch vehicles, and one-shot missions. It was envisioned that considerable savings in money and manpower could be realized in carrying out R&D support programs with a "general-utility" spacecraft of this nature. It was felt that no new techniques would be necessary to develop the hardware for such a spacecraft and during the development of the first few units a set of standard off-the-shelf modules would become available for future missions.

After detailed examination of the characteristics and requirements of a large inventory of experiments from the ARSP and SESP, it was determined that it was virtually impossible to develop a spacecraft with a single basic configuration to adequately meet the needs of all the experiments, let alone the constraints of the various launch vehicles and TT&C ranges required to support the experiments. It became apparent that several spacecraft with various payloads, volume, weight, power, and attitude-control capabilities would be required. In addition, if "rides of opportunity" and primary payload space on a variety of launch vehicles were to be utilized effectively, spacecraft of several overall sizes from small (15 to 30 lb total) to large (>500 lb total) with minimum launch vehicle interfaces would be required. Preliminary feasibility studies aimed at defining a new "family" of spacecraft to meet these requirements were carried out. The estimated initial development costs for the resultant designs were prohibitively high for the limited funds available to the R&D support type program. As a result, the new general-utility spacecraft family concept was abandoned in favor of a concept which would avoid initial hardware development costs where possible. This concept centered on the direct use or modification of existing off-the-shelf spacecraft components and subsystems to synthesize in "tinker toy" fashion the required spacecraft.

B. Hardware

To initiate the development of this concept, an industry-wide survey[1] was conducted in late 1965 to gather detailed technical information on previously developed spacecraft that could be adapted as general-utility space test platforms. The survey was designed to provide information which would permit cataloging of existing spacecraft by configuration, subsystem characteristics, adaptability as general-utility space test platforms, previous orbital history, and estimated cost breakdown per unit if launched on the vehicle for which the spacecraft was originally designed.

The survey indicated that many spacecraft could be adapted, that a variety of subsystems were readily available, and that the new general-utility spacecraft development was definitely not warranted. Those spacecraft (22) and versions thereof (50+) that were reviewed and deemed suitable for adaptation as general-utility space test platforms are listed in Table 1, along with their approximate gross weight, fabrication lead time, and manufacturer. The spacecraft that were deemed not suitable are listed in Table 2.

To allow a more detailed evaluation of the general-utility spacecraft family concept, a brief description of the current configuration of each spacecraft in the OV family (OV1, 2, 3, and 5) and their growth potentials are presented in the Appendix. These spacecraft are sponsored by the OAR for implementation of the ARSP. They were specifically developed as general-utility vehicles utilizing off-the-shelf hardware to a maximum extent. The utilization of this hardware often requires the experimenter to relax experiment (or test) requirements.[2] This seldom results in unsatisfactory compromises in the experiment, and it yields significant cost savings. These savings are realized not only by the use of the proven hardware, but also be the resulting minimization of associated software (documentation, quality control, and reliability) and environmental test programs. The general-utility nature of these spacecraft is aptly illustrated in Table 3,[3] which shows the variety of experiments orbited by the OV1 system.

Table 1. Developed Spacecraft Suitable for Use or Modification as General-Utility Spacecraft

SPACECRAFT	VERSIONS AVAILABLE	MAX APPROX GROSS WT (lb)†	APPROX LEAD TIME (mo)††	MANUFACTURER
TRS - Tetrahedral Research Satellite	6	5 to 12	5	TRW
ORS (OV5) - Octahedral Research Satellite	9	7 to 45	5	TRW
SECOR II - Sequential Collation of Range	1	45	6	Cubic or ITT
SECOR I - Sequential Collation of Range	1	55	6	Cubic or ITT
TIROS (24 in. baseplate) - Television & Infrared Observations Satellite	1	105	12	RCA
GGTS - Gravity Gradient Test Satellite	1	125	11	GE
BUS - Bendix Utility Satellite	1	145	11	Bendix
OV3 - Orbiting Vehicle Type Three (ARSP)	3+*	205	11	Aerojet (SGD)
TIROS/TOS - Television & Infrared Observations Satellite/TIROS Operational Satellite	1	300	11	RCA
OV1 - Orbiting Vehicle Type One (ARSP)	3+*	330	11	Convair (GDC)
ARS - Apollo Range Satellite	1	400**	11	Hughes
TOS-APT/TR - TIROS Operational Satellite Automatic Picture Transmission/Tape Recorder	1	425	12	RCA
OV2 - Orbiting Vehicle Type Two (ARSP)	3+*	450	11	Northrop (NSL)
VELA	2	530	20	TRW
OSO - Orbiting Solar Observatory	1	690	20	Ball Brothers
OGO - Orbiting Geophysical Observatory	1	1150	26	TRW
NIMBUS	3*	1200	21	GE
BIOSAT - Biological Satellite	3	1265**	14	GE
ATS - Applications Technology Satellite	3	1550**	14	Hughes
OAO - Orbiting Astronomical Observatory	2	4040	36	Grumman
BURNER II	2	1735Δ	9	Boeing
OV1 PROPULSION MODULE	2	883ΔΔ	11	Convair (GDC)
TOTALS 22	50+	--	--	13

† Payload capability can be crudely approximated by dividing the gross weight by 2.4. Contact manufacturers for accuracte figures.

†† As of early 1966. Contact manufacturer for accurate estimates for specific missions. Defined as contract go-ahead to delivery as a complete integrated flight unit.

* Easily varied solar power capability.

** Total qualification weight. Includes solid-propellant motor that can be replaced with experiments.

Δ Includes 1440-lb solid-propellant motor. Can be converted to 3-axis stabilized platform with 4000-lb payload capability.

ΔΔ Includes 605-lb solid-propellant motor. Can be converted to 3-axis stabilized platform with 437-lb payload capability.

Table 2. Developed Spacecraft Not Suitable for Use or Modification as General-Utility Spacecraft

SPACECRAFT	VERSIONS AVAILABLE	REASON NOT SUITABLE
OSCAR - Orbiting Satellite Carrying Amateur Radio	Several	Similar to ORS but larger - no specific manufacturer
SOLRAD - Solar Radiation Satellite	Several	Similar to SECOR I
AOSO - Advanced Orbiting Solar Observatory	1	Development contract cancelled by NASA
EXPLORER Series	Several	Similar to OV1, OV3, BUS, and GGTS
CENTAUR Upper Stage	1	Under development
OWL	1	Under development - similar to OV1 and OV3
S^3 - Small Standard Satellite	Several	Under development - similar to GGTS, OV1 OV3, and TIROS (24-in. baseplate)
LUNAR ORBITER	1	Relatively high cost - long lead time - similar to NIMBUS
GREB - Galactic Radiation Experimental Background Satellite	Several	Special purpose
GGSE - Gravity Gradient Stabilization Experiment	Several	Special purpose
IMP - Interplanetary Monitoring Probe	Several	Special purpose - no specific manufacturer
LES - Lincoln Experimental Satellite	Several	Special purpose - no specific manufacturer
TRANSIT	1	Not adaptable
RELAY	1	Not adaptable
SYNCOM - Synchronous Communication Satellite	2	Special purpose - similar to OV1 and OV3
TIROS (30-in. baseplate) - Television & Infrared Observations Satellite	2	Special purpose - similar to OV1 and OV3
PIONEER	1	Special purpose - similar to OV1 and OV3
SMS I - Solar Monitor Satellite I	1	Proposal only
SMS II - Solar Monitor Satellite II	1	Proposal only
GASP - Gravity Anchored Sun Pointed Satellite	1	Proposal only
POEM - Polar Orbiting Earth Monitor	2	Proposal only
POSM - Polar Orbiting Solar Monitor	1	Proposal only
SPARES - Space Research Satellite	Several	
PEGASUS	1	Relatively high cost - special purpose
SURVEYOR	1	Relatively high cost - special purpose
RANGER 1 through 5	2	Relatively high cost - long lead time - no specific manufacturer
RANGER 3 through 9	1	Relatively high cost - long lead time - no specific manufacturer
MARINER	2	Relatively high cost - long lead time - no specific manufacturer
MERCURY	1	Relatively high cost - special purpose - man rated - long lead time
GEMINI	1	Relatively high cost - special purpose - man rated - long lead time
APOLLO	1	Relatively high cost - special purpose - man rated - long lead time

Details on the adaptability of other spacecraft identified in Table 1 are not presented due to the limited scope of this paper and, in some instances, the proprietary nature of the information. Interested individuals or agencies can obtain these details by contacting individual manufacturers.

C. Cost

The hardware costs(1) of the spacecraft listed in Table 1 are generalized in Fig. 1. The hardware included in the curves are the structure, data handling, telemetry, tracking, command, electrical power, temperature control, stabilization and orientation, engineering status and proplusion systems. The weight associated with the propulsion system does not include propellant weight. Payload weights and costs are also not reflected. The data are based on replicas of the spacecraft as originally configured.

Although the curves do not reflect all costs associated with a spacecraft program, they do indicate a relatively low cost associated with the OV general-utility spacecraft family.

To the costs of Fig. 1, recurring software, environmental test, aerospace ground equipment (AGE), assembly and checkout, payload and payload integration, and flight support, as well as launch vehicle and launch vehicle integration costs, need to be added. Of these, all but the recurring software costs are missions-peculiar and difficult to generalize. However, survey results indicate that the recurring software costs can vary from 15% ($777/lb) to 30% ($1520/lb) of the total spacecraft recurring hardware costs, depending on the type of software programs imposed.(4) The $777/lb figure corresponds to a minimum OV type low-cost program under which the contractor uses his own documentation and reliability control system and MIL-I-45208A, "Inspection System Requirements," as a minimum quality control program. The $1520/lb figure corresponds to a program which imposes AFSCM 310-1, "Management of Contractor Data and Reports;" MIL-STD-785, "Requirements for Reliability Program (for Systems and Equipment);" and MIL-Q-9858A, "Quality Program Requirements," or their equivalents.

Table 3. OVI Experiments in Orbit

TITLE	AGENCY
Measurement of Magnetospheric Electric Fields	NASA Goddard Space Flight Center
Orbiting Algae Systems	AF School of Aerospace Medicine
Thermal Control Coatings	AF Materials Lab.
Bio-Hazards Associated with Space Radiation	AF Weapons Lab.
Verification of Mathematical Shielding Models	
Reflective Open Grid Passive Radar Studies	AF Research & Technology Div.
Thermal Control Coatings	
Spinning Spacecraft Attitude Determination System	AF Cambridge Research Lab.
Spacecraft Magnetic Field Measurements	
Heavy Primary Cosmic Ray Telescope	
Background Radiation	
Cosmic Radiation	
Exospheric Radiation	
All-Sky Lyman-Alpha Photometer	Aerospace Corp. Space Physics Lab.
UV Dayglow Photometry	
Multicolor Nightglow Photometry	
Solar X-Ray Spectrometer	
Omnidirectional Proton Spectrometers	

D. Management

The success of general-utility spacecraft programs depends heavily on the management philosophy and structure of both the procuring agency and the contractor. A rigid operating philosophy must be established prior to the commencement of a program. The procuring agency must know in detail what is required as a contract end item and clearly specify it in both the request for proposal (RFP) and the final work statement. The contractor must fully understand the work statement at the onset. A rapport of mutual trust and respect must be developed between the personnel of both organizations. During all interactions, both the contractor and the procuring agency personnel must be sensitive to situations which could disrupt this rapport. The contractor must be given as much free rein as possible.

Consistent with a low-cost program structure, the cognizant or project-engineer type organization is recommended for both the procuring agency and the contractor. Under this concept, the project is subdivided into various subsystem tasks and each engineer is delegated full responsibility for his assigned subsystem. This approach tends to reduce project personnel to a minimum, yielding a streamlined organization and maximum personal rewards for those responsible for the various portions of the project. This is an extremely important, yet often neglected, point.

A test program which satisfies the criteria of integrity assurance at a minimum cost must be established. Structures can usually be qualified by similarity to the structure developed during the original program, eliminating the need for a qualification or proof test model. The qualification or proof test structure from the original program is usually available and can be modified as required for the thermal test, EMI, and launch vehicle fit check models. Final readiness is demonstrated for the flight unit by a series of acceptance level environmental and functional tests. The environmental tests usually needed are thermal vacuum and random vibration environment. For simplicity, the random vibration environment should be induced by acoustic input in a reverberation chamber to levels and durations equal to those expected during launch.

The procuring agency must be aware of and make maximum use of available government furnished equipment (GFE) from previous programs as well as maximum use of government facilities. For example, the solar cell modules of the three OV2 spacecraft fabricated to date were GFE supplied from the Advanced Research Projects Agency (ARPA) ARENTS program cancelled in 1963. The solar cell modules and the OV2-5 thermal test model were tested in the USAF Arnold Engineering Development Center (AEDC) solar simulator facility by AEDC personnel. The use of the GFE and government facilities must be specified in both the RFP and final work statement.

E. Summary

The application of the above-stated principles, utilization of off-the-shelf hardware, minimization of recurring hardware and software costs, well defined RFP's and work statements, close-knit but flexible management, and maximum use of GFE and government facilities are the keys to the cost-effectiveness of the general-utility spacecraft family concept.

The general-utility OV spacecraft family demonstrates that spacecraft manufacturers, hardware suppliers, as well as the procuring agencies in general, are now mature enough in program management and spacecraft technology to reduce the high costs associated with many past programs. From the current results of the OV programs, it appears more expedient and cost-effective in many

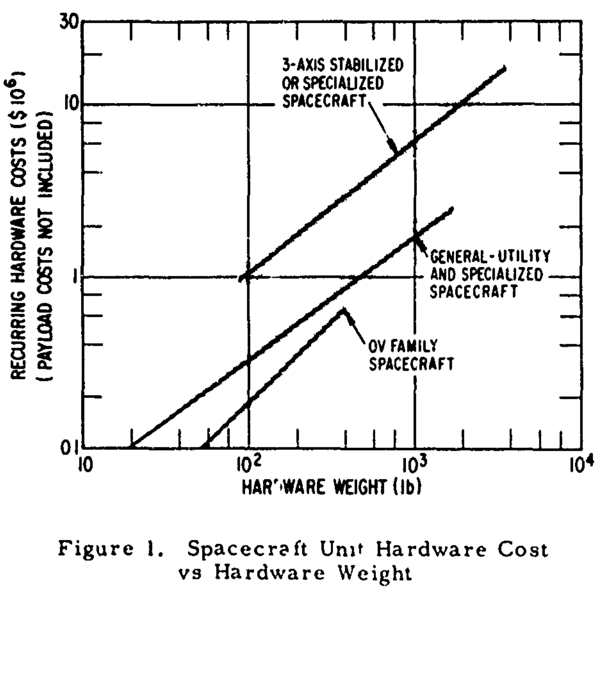


Figure 1. Spacecraft Unit Hardware Cost vs Hardware Weight

cases to use several "desophisticated" general-utility spacecraft and minimize recurring hardware and software costs and fabrication lead times, as well as the formidable task of payload integration, as opposed to using the special-purpose or large observatory type spacecraft.

III. Orbital Bus Concept

A. History

If general-utility spacecraft are to be used effectively, methods are required to provide them with orbit transfer capabilities independent of the primary payload launch vehicle. The integrated single-burn "kick motor" concept has been used on a variety of specially designed spacecraft such as the SYNCOM and the Applications Technology Satellites (ATS). The disadvantage of this concept is that the motor is internally integrated into the structure, which generally makes a change of motors (enlargement) difficult without major modification of the structure. The advantage of the concept is that the spacecraft generally assumes a stable configuration in which the thrust axis of the motor is the spin axis of the spacecraft, which, by necessity, must be the major moment-of-inertia axis if stable on-orbit orientation is to be achieved following motor burnout.

Unfortunately, the majority of available general-utility spacecraft do not lend themselves to the integrated kick motor approach. However, most of these spacecraft are designed for spin stabilization and can be readily adapted to the external integration of a single motor. This provides them the capability of transferring from elliptical to elliptical, elliptical to circular, or circular to elliptical orbits. However, in many instances it would be desirable to tranfer from an elliptical to elliptical orbit where both the apogee and perigee of the final orbit are lower or higher than those of the initial orbit or from circular to lower or higher circular orbits. When these orbits are not the final orbit of the launch vehicle upper stage, a dual-burn propulsion capability is required on the spacecraft.

To adapt various members of the general-utility spacecraft family to these requirements, the "orbital bus" concept has evolved. The concept involves a modular, low cost, and somewhat radical approach to the problem and centers around a maximum utilization of off-the-shelf hardware. The use of orbital buses on multiple-payload launch missions eliminates the need for several launch vehicles to accomplish the missions of spacecraft with significantly different orbits. These features make the concept unusually cost-effective.

B. Configurations

An orbital bus consists of several basic elements: appropriate solid-propellant rocket motors, a stabilization system, a power and control system, a launch vehicle adapter and separation system, and an appropriate structure which accommodates all elements and one or several general-utility spacecraft, all assembled in a modular form. These elements in a representative configuration are illustrated in Fig. 2, along with a typical multiple-orbit/payload mission flight sequence. The orbital bus provides the propulsion, logic, and time sequencing necessary to transfer the spacecraft from the initial to the final orbit. To achieve this, two thrusting periods or impulses are required. The inertial directions of these two impulses are 180 deg apart; thus, the rocket motor nozzles must be separated by 180 deg. Because of the constraint of using available general-utility spacecraft and packaging limitations usually inherent in integrating multiple payloads on launch vehicles, an elongated configuration as illustrated usually evolves. This simple modular configuration is the most reliable, least complex, and most cost-effective. Such configurations are inherently unstable since the spin (longitudinal) axis is not the major moment-of-inertia axis and since structures are not perfectly rigid.

C. Stabilization System

The inherent difficulty with the unstable bus configuration is the divergence of the precession cone angle during coasting phases. The coning at the first and second burns will reduce the velocity gained, thereby inducing final orbit dispersions. Factors contributing to the dispersions are: parent vehicle attitude and rate errors, tipoff errors induced by separation from the parent vehicle, spin-up motor nozzle misalignment and unequal thrust, main motor misalignment, main motor thrust tolerances, timer errors, and structural damping. During coast phases, any residual coning will diverge for an unstable configuration. This divergence is a function of the spin rate, the coast time, and the energy dissipation due to damping within the vehicle structure. Appropriate optimization of these parameters is essential to retain relatively small coning angles. The coning, as such, does not cause orbit dispersions until motor ignition.

Possible stabilization techniques for use on buses with unstable configurations are: spin about the major moment-of-inertia axis achieved by deploying booms, utilization of a momentum wheel or fluid flywheel, active deconing with cold gas jets, and a passive spin-despin system with a pendulum damper.[5] A summary of the salient features of each concept is presented in Table 4. Where long transfer periods (>2 hr) and accurate final orbits are necessary, the momentum wheel should be used. Such wheels can be procured off-the-shelf and have been used for stabilization on ballistic probe flights and in control applications on spacecraft such as the Orbiting Geophysical Observatory (OGO). The wheels are available in various sizes and their rotational speed can be varied to accommodate a large range of angular momentum requirements. The spin axis of the wheel need only be parallel to the spin axis of the bus. Usually better packaging can be obtained by mounting the wheel off of the spin axis of the bus. Tipoff errors caused by separation of the bus from the parent vehicle can be virtually eliminated by spinning up the wheel while still on the parent vehicle. The spin rate of the wheel can be used as a separation command enable signal. This will assure a stabilized bus and preclude possible collisions with the parent vehicle due to an unstable bus at the time of motor ignition.

A unique application of the momentum wheel to an orbital bus configuration is shown in Fig. 3. The launch vehicle separation and remaining sequence of events for this bus are illustrated in Fig. 4. The general sequence of events is applicable to any bus employing a momentum wheel for stabilization except that the first burn motor is

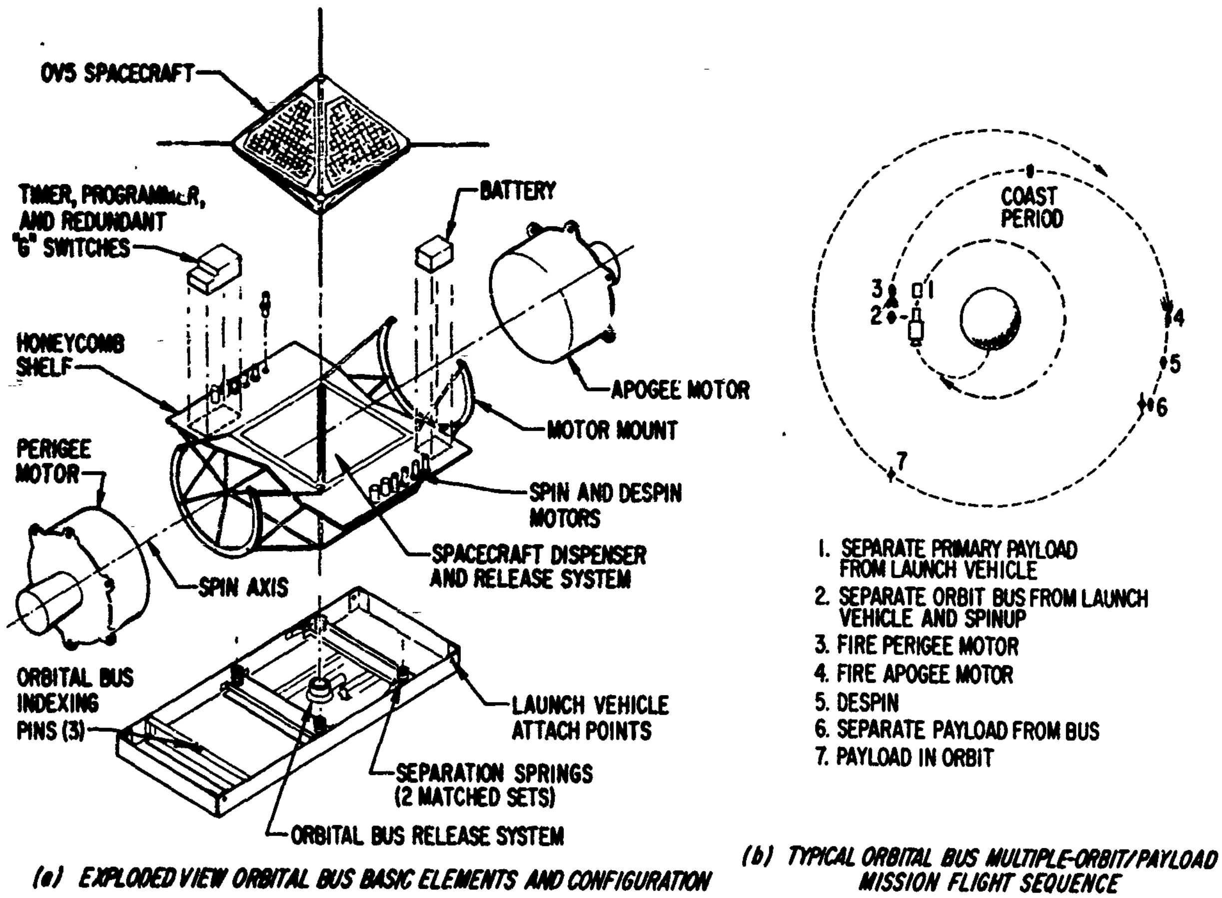


Figure 2. The Orbital Bus Concept

Table 4. Summary of Stabilization Concepts

SPIN ONLY • Unstable • Poor orbit accuracy • Simple • Low cost	BOOMS • Stable • Good orbit accuracy • High weight penalty • Two-stage spinup • Despin difficulties • Development required • Moderate cost
REACTION WHEEL • Stable • Best orbit accuracy • Moderate weight penalty • No spin-despin rockets • Off-the-shelf • Moderate cost	FLUID FLYWHEEL • Stable • Best orbit accuracy • Moderate weight penalty • No spin-despin rockets • Long development time • High cost
ACTIVE DECONING • Stable • Good orbit accuracy • Low weight penalty • Relatively poor reliability • Development required • Moderate cost	SPIN-DESPIN • Stable • Good orbit accuracy • Low weight penalty • Long development time • High cost

not usually separated from the bus when the spacecraft which it carries are not an integral part of the structure as shown in Fig. 5.

D. Propulsion System

The solid-propellant rocket motors are constrained to off-the-shelf units with burn times in excess of 10 sec to minimize cost and the acceleration during burn. The motors can usually be off-loaded to provide a wide velocity increment (ΔV) capability. The technique of pyrotechnically removing the nozzle from a motor to achieve thrust termination at a given ΔV for specific motors is not recommended because of possible adverse effects on the bus stability. It has been found that motors of a desired specific propellant load are readily available as off-the-shelf units although gaps exist in the 25- to 40-lb, 280- to 490-lb, and 600- to 900-lb propellant load regions. In many cases, two identical motors can be used by adjusting the positions of the burn of the first (perigee) motor in the initial orbit and the burn of the second (apogee) motor in the transfer orbit.

E. Power and Control System

Power for the various subsystems on the bus is best provided by a sealed primary Ag-Zn storage battery.[6] The sequence of events can be controlled by a timer/programmer, although ground command can be used for configurations as shown in Fig. 3. A solid-state magnetic logic timer/programmer, which does not reset in the event of rfi transients or either short or long term power dropouts, is recommended.[7][8] Ideally, the unit should be capable of being programmed in the field. At least one such timer/programmer exists as off-the-shelf hardware.

F. Launch Vehicle Separation System

A launch vehicle adapter and separation sys-

tem is required to reduced the launch vehicle interface to a mechanical bolt-on operation. Typical mechanisms are illustrated in Figs. 2 and 3. The mechanism should be fitted with a battery, timer, and redundant g-switches intended to provide a self-contained separation signal for the bus, thus eliminating all electrical interface with the launch vehicle. The g-switches should be single-event type which close at launch vehicle liftoff.

IV. Multiple-Orbit/Payload Launch Concept

A. History

The concept of the multiple payload launch dates back to the successful Transit 2A/SOLRAD 1 mission of 22 June 1960.(9) The multiple-orbit/payload launch concept, a step beyond the multiple payload concept, originated with the USSR Venus 1/ Sputnik 8 program of 12 Feb 1961.(9) The most spectacular launch of this nature was the Titan IIIC-9 ARSP/HST flight in Nov 1966, which orbited three spacecraft carrying a total of 1800 lb of experiments and provided a semiballistic trajectory for the qualification test of a reentry heat shield capsule. The cost-effective potential of this type of launch has not been fully exploited as yet.

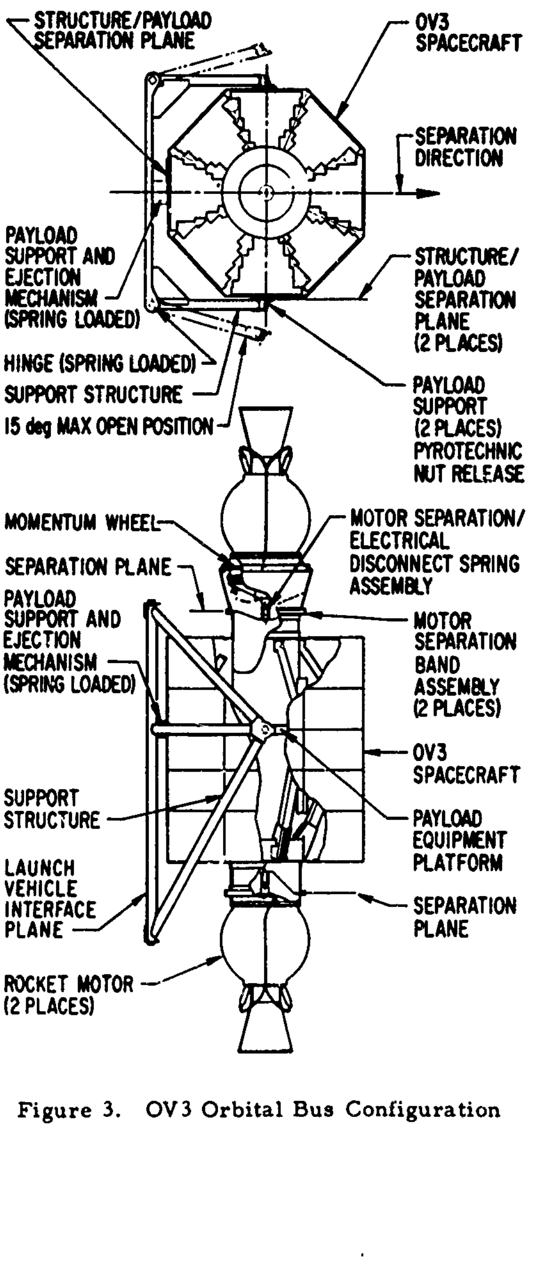


Figure 3. OV3 Orbital Bus Configuration

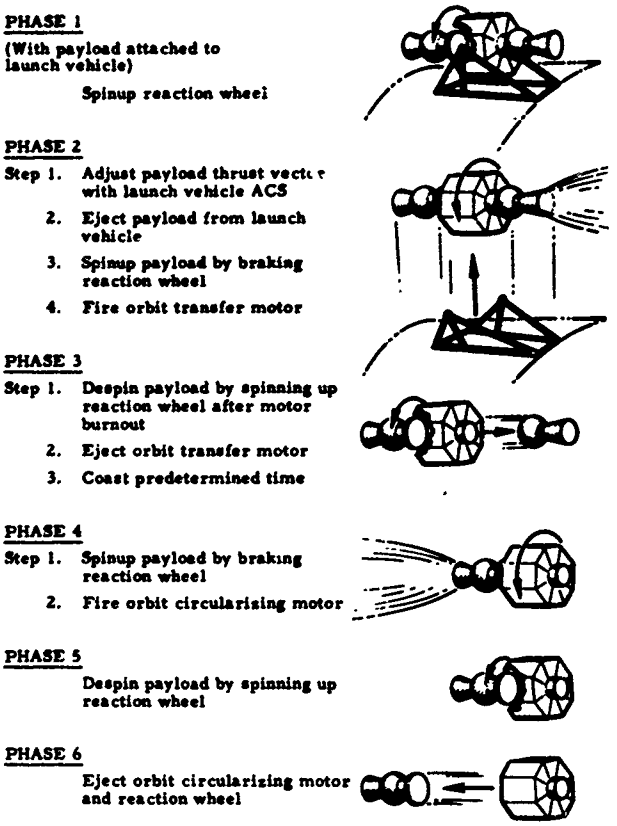


Figure 4. OV3 Orbital Bus Event Sequence (typical for all momentum wheel augmented buses)

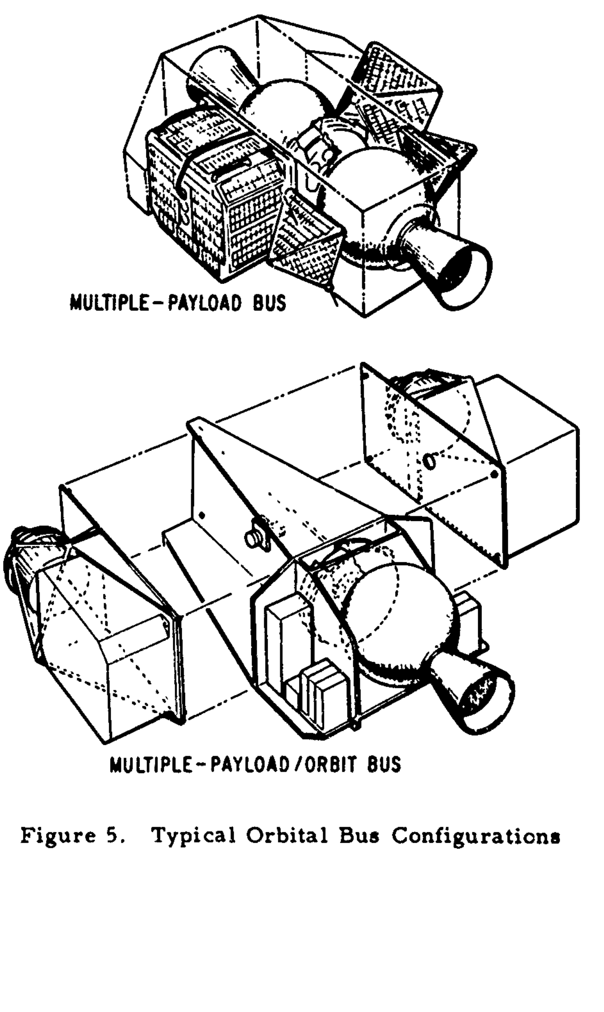


Figure 5. Typical Orbital Bus Configurations

B. Definition

The multiple-orbit/payload launch is a mission concept in which several spacecraft, combined as necessary with kick motors and orbital buses, are integrated on a single launch vehicle. The concept encompasses single and multi-burn launch vehicle final stages, as well as the conversion of the final stages to general-utility space test platforms by the integration of experiments requiring short (several hours to several days) on-orbit lifetimes directly on the stages. The concept centers on the fact that maximum cost-effectiveness is gained for a launch vehicle when all its payload capability for a given mission is completely utilized. However, the minimum-capability, single launch vehicle able to accomplish a specific mission is not necessarily the most cost-effective. Instances occur for specific missions in which it is more economical to purchase two lower capability vehicles as opposed to one which will just meet the needs of the payload. These situations occur due to launch vehicle and launch pad availabilities and differences in vehicle reliabilities.

This section will discuss several aspects of multiple-orbit/payload missions. Items such as experiment, spacecraft, launch vehicle, launch range, and tracking range information collection methods, experiment/spacecraft and spacecraft/launch vehicle integration techniques, mission planning analysis and reliability, and cost-effectiveness analysis are not covered. These are complex subjects worthy of papers in themselves.

C. Cost-Effective Program Planning

The usual measure of "cost-effectiveness" is the cost effectiveness index (CEI) defined as:

$$CEI = \frac{\text{(total mission costs)}}{\text{(weight in orbit)(reliability)}} \left(\frac{\$}{lb}\right)$$

In this expression the total mission costs include the cost of the launch vehicle and its launch, the spacecraft which it carries, experiment and spacecraft integration on the launch vehicle, and experiment integration in the spacecraft. This does not include the cost of on-orbit support, data analysis or reduction, and data publications, since such quantities cannot be realistically tied to a reliability figure. The weight in orbit consists only of the payload separated from the launch vehicle and any payload hard-mounted to the final stage. Structure, batteries, solar panels, telemetry, thermal control systems, etc., which have been added to the stage for experiment support, are considered a part of the weight in orbit. The launch vehicle includes all kick motors, orbital buses, payload support structures, and final-stage-retained spacecraft separation mechanisms, i.e., everything required to put the experiments and their on-orbit support systems into the required orbits. Since it is usually not necessary to meet a narrow launch window in missions of the type under discussion, the reliability term includes only the reliability of the launch vehicle from the time of motor ignition.

The CEI used in the following program planning example is a simplification of the above definition. Since the reliabilities of the launch vehicles used are approximately equal, this term was eliminated from the CEI calculations. Thus, the CEI's indicated are straightforward measures of the dollars per pound needed to put the payload in the required orbits independent of the launch vehicle reliabilities.

This example illustrates the improved cost-effectiveness that can be realized through the use of multiple-orbit/payload missions. The program involves the flight of four independent spacecraft, each requiring different orbits and belonging to separate agencies. The requirements and characteristics of these spacecraft pertinent to their launch are summarized in Table 5. It is assumed that all spacecraft have approximately equal DOD priority and that each agency can justify the cost of procuring their own individual launch vehicle.

If each agency acts independently of the others, as often occurs, four Scout (SLV-1A) launch vehicles would be required to carry out the program (i.e., fly all spacecraft). The CEI for the program is \$23,600/lb as summarized in Table 6, Approach "A." Note that spacecraft 1 was injected into an initial elliptical orbit prior to achieving its final orbit. A direct trajectory was not used since it would degrade overall mission reliability even though the payload weight capability to the final orbit would be greater. This statement is made since the spacecraft is capable of obtaining usable data in the 2400 x 500 n mi transfer orbit. This fact usually applies to most R&D type spacecraft. If a direct orbit injection were used and injection stage malfunctioned, no usable data could be obtained since the stage and spacecraft would be in a ballistic trajectory. This point is often not considered in mission planning. Note also that the desired nominal orbits of all spacecraft were achieved but that all available launch vehicle capability was not used. If this capability could be entirely used, the program CEI would be improved. The difficulties in acquiring the use of this capability for secondary payloads by agencies other than the agency buying the vehicle are many. In most cases it never happens. Cost-effective utilization of secondary payload capability has not been fully achieved in the past. The use of multiple-orbit/payload launches is aimed at eliminating this situation.

Table 6, Approach "B," presents a mission configuration in which agencies X and Y jointly procure a single Scout and fly spacecraft 2 and 3 in a multiple-orbit/payload launch. This step reduces the program CEI to \$19,500/lb, which is a 17.4% improvement over Approach "A." The calculations include the increased cost of spacecraft-to-launch-vehicle integration caused by the multiple payload. Note the reduction in the unused launch vehicle capability from a total of 210 to 120 lb.

In Approach "C," agencies X, Y, and Z jointly procure a single launch vehicle to replace two of the Scouts. This move further reduces the program CEI to a value of from \$14,000/lb to \$17,300/lb. The lower figure corresponds to an agency launch and the higher to a contractor launch. Assuming an agency launch, this is a 40.6% improvement over Approach "A." Note the increase in excess payload capability because of the higher performance of the Thor/Burner II over the two Scouts, as well as the acceptable compromise (see Table 5) in the final orbits of spacecraft 2 and 3

Table 5. Example Program Spacecraft Requirements and Characteristics Summary

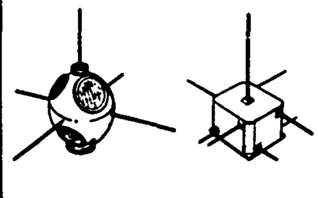
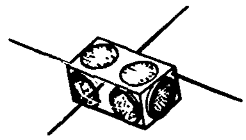
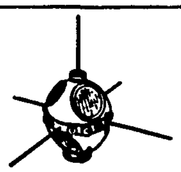
SPACE-CRAFT	CONFIGURATION	OWNING AGENCY	ORBIT: ALTITUDE (n mi)	ORBIT: INCLINATION (deg)	WEIGHT (lb)	DESIRED LAUNCH DATE	SPECIAL REQUIREMENTS
1	Spacecraft 1 and 2 are identical in configuration. Each may have one of two configurations designated I and II, defined below. Configuration I is preferred.	X	2400^{+100}_{-400} nominally circular	90^{+0}_{-15}	Type I 55 Type II 45	Sept ± 1 mo	Final tumble rate <3 rpm about any axis. Type I can be equipped with a despin yo-yo. Type II has no despin capability.
2	 TYPE I TYPE II	X	800^{+200}_{-100} nominally circular	90^{+0}_{-15}		Nov ± 1 mo	
3		Y	150^{+50}_{-0} 800^{+400}_{-100} elliptical	90±15	60	ASAP but no later than December	Final tumble rate <1 rpm about any axis. No despin capability.
4		Z	600±200 nominally circular	75±15	120	ASAP but no later than December	Final tumble rate <3 rpm about any axis. Equipped with despin yo-yo.

Table 6. Example Program Summary (all orbit inclinations ≈90 deg)

APPROACH AND DEFINITION	SPACECRAFT (S/C)	LAUNCH VEHICLE (L/V)	KICK STAGE TYPE	L/V INITIAL ORBIT ALT (n mi)	S/C FINAL ORBIT ALT (n mi)	APPROX EXCESS CAPABILITY (lb)	COST ($k/lb)	PROGRAM COST ($k/lb P/L)
"A" All individual launches	1 (Type I)	Scout 1	TE-M-458	2400 × 500E	2400C	0	33.6	23.6*
	2 (Type I)	Scout 2	None	800C	800C	50	30.0	
	3	Scout 3	None	800 × 150E	800 × 150E	190	28.4	
	4	Scout 4	None	600C	600C	20	14.6	
"B" Two individual with one multiple orbit/payload launch	1 (Type I)	Scout 1	TE-M-458	2400 × 500E	2400C	0	33.6	19.5*
	2 (Type II)	Scout 2	LPC-2P 14102-9	800 × 150E	800C	100	17.8	
	3		None		800 × 150E			
	4	Scout 3	None	600C	600C	20	14.6	
"C" One individual with one multiple orbit/payload launch	1 (Type I)	Scout 1	TE-M-458	2400 × 500E	2400C	0	33.6	
	2 (Type II)	Thor (SLV-2)/ Burner II	LPC-2P 14102-9	700 × 150E	700C	140	9.4** or 13.0**	14.0* or 17.3**
	3		None		700 × 150E			
	4		LPC-2P 14102-1		700C			
"D" One multiple orbit/payload launch involving one orbital bus	1 (Type II)	TAT (SLV-2A)/ Burner II	2 TE-M-458 in orbital bus	700 × 150E	2400C	0	See next col	10.51* or 13.55**
	2 (Type II)		LPC-2P 14102-9		700C			
	3		None		700 × 150E			
	4		LPC-2P 14102-1		700C			

E = Elliptical   C = Circular   *Agency-launched   **Contractor-launched

which allowed flight of spacecraft 4 on the same launch without adding significantly to the mission complexity. Such compromises are characteristic of multiple-orbit/payload missions.

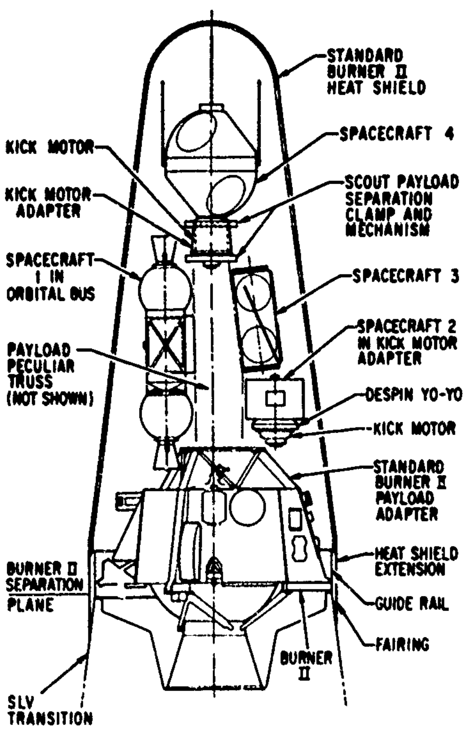
An alternate configuration to this launch is to integrate directly on the Burner II a single rocket motor 180 deg from the nozzle of the Burner II motor and eliminate the individual kick motors on spacecraft 2 and 4. Spacecraft 3 would separate from the Burner II prior to firing of the single kick motor which would circularize the Burner II and spacecraft 2 and 4 in their final orbits. This would reduce the excess payload capability and increase the overall mission reliability. The salient point is that both Approach "C" configurations yield as much CEI improvement as possible based on current launch concepts; this is due to the divergence of the orbits between spacecraft 2, 3, and 4 and spacecraft 1.

However, at this point, the orbital bus concept developed in the previous section allows still further CEI improvements (Table 6, Approach "D"). The substitution of an orbital bus for the Scout to accomplish the mission of spacecraft 1 necessitates the choice of a higher performance launch vehicle but eliminates all excess capability, thus allowing maximum cost-effectiveness to be achieved. The CEI achieved is $10,510/lb to $13,550/lb, depending on an agency or contractor launch. Assuming an agency launch, this amounts to a 55.5% improvement over Approach "A" and a 24.9% improvement over Approach "C," the best configuration that can be achieved without the orbital bus. The alternate Approach "C" configuration was not used in this launch since it could not physically be implemented because of limited launch vehicle capability. This is due to the fact that the motor used to provide the necessary delta velocity required to circularize the Burner II and spacecraft 2 and 4 weighs more than the two small motors required to circularize only the two spacecraft.

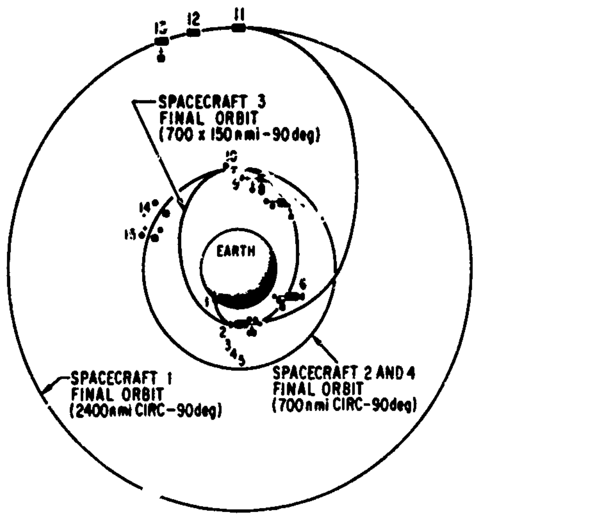
The packaging of the spacecraft on the launch vehicle and the mission profile for Approach "D" are illustrated in Fig. 6.

D. Atlas Applications[10]

Atlas (SM-65) D, E, and F series launch vehicles have been and currently are used by the OAR with the OV1 spacecraft and its propulsion module (P/M) to achieve multiple-orbit payload



(a) LAUNCH VEHICLE INTEGRATION LAYOUT



	EVENT SEQUENCE	TIME (min)
1	LIFT-OFF	0
2	INJECT BURNER II IN 90 deg, 150 x 700 nmi ORBIT	5.9
3	SEPARATE ORBITAL BUS (SPACECRAFT 1)	7.5
4	TURN BURNER II AROUND	8.5
5	FIRE ORBITAL BUS PERIGEE MOTOR	8.7
6	SEPARATE SPACECRAFT 3 (ΔV ≈ 1 ft/sec)	10.0
7	ORIENT BURNER II AND SPIN TO ≈ 75 RPM WITH AUGMENTED STABILIZATION SYSTEM	51.0
8	SEPARATE SPACECRAFT 2 (ΔV ≈ 6 ft/sec)	52.0
9	SEPARATE SPACECRAFT 4 (ΔV ≈ 12 ft/sec)	53.0
10	SPACECRAFTS 2 AND 4 CIRCULARIZATION KICK MOTORS AUTOMATICALLY FIRE	56.2
11	ORBITAL BUS AUTOMATICALLY FIRES CIRCULARIZATION MOTOR	73.9
12	ORBITAL BUS AUTOMATICALLY DESPINS	75.0
13	SPACECRAFT 1 AUTOMATICALLY EJECTS FROM ORBITAL BUS	75.5
14	SPACECRAFT 2 AND 4 AUTOMATICALLY DESPIN	87.0
15	SPACECRAFTS 2 AND 4 AUTOMATICALLY EJECT CIRCULARIZATION KICK MOTORS	07 5

(b) FLIGHT PROFILE

Figure 6. Example Program TAT/Burner II Multiple-Orbit/Payload Mission

missions. The missions are accomplished with dual OV1 system installations on the nose of the Atlas (Fig. 7). A third OV1 can be side-mounted on the Atlas for a three-in-one mission using a coffin-like structure (Fig. 8).

The OV1's can be injected into circular or elliptical orbits. On Atlas flights the OV1 system separates from the booster shortly after sustainer engine cutoff (SECO) by sensing the termination of acceleration. The known attitude of the Atlas is used as a reference by the OV1 guidance and attitude control (GAC) subsystem. The OV1 then coasts in a ballistic trajectory while performing programmed pitch and roll maneuvers to achieve the required attitude for firing the P/M. The GAC system maintains vehicle orientation during burn.

After orbit injection the P/M maintains its attitude until spacecraft separation, which occurs a short time after motor burnout. At this point, power to all P/M components is turned off, except to the telemetry rf carrier and a C-band radar beacon which remain on for downrange tracking and ephemeris determination until battery depletion.

The Atlas/OV1 P/M performance capabilities are shown in Fig. 9.[11] Trajectory shaping can be used to achieve a variety of orbits. Figure 9 indicates a minimum altitude of 740 n mi for a 400-lb payload; however, the Atlas/OV1 system can be targeted to provide lower circular orbits. Typical circular orbits flown are 250 to 500 n mi.

The Atlas/OV1 P/M combination need not be used with the OV1 spacecraft. Orbital buses similar to those shown in Fig. 5 can be adapted to the

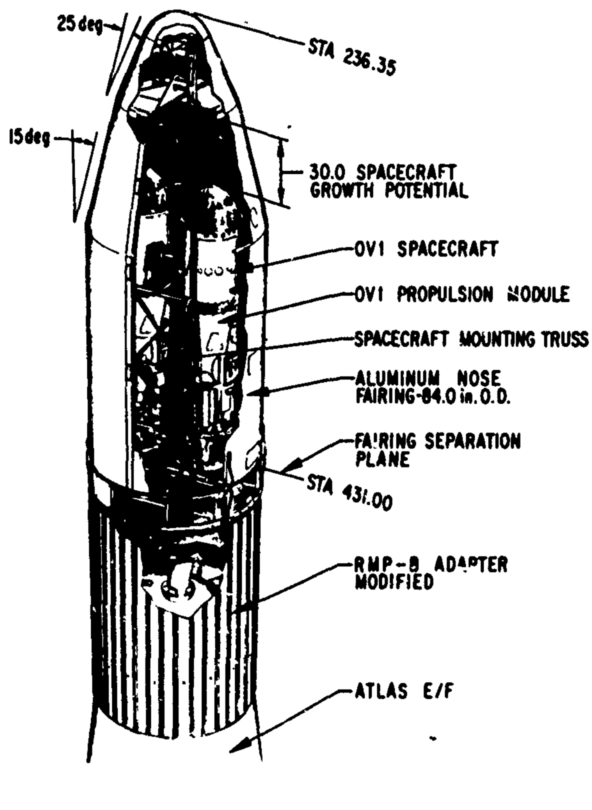


Figure 7. Atlas E/F Dual OV1 Installation

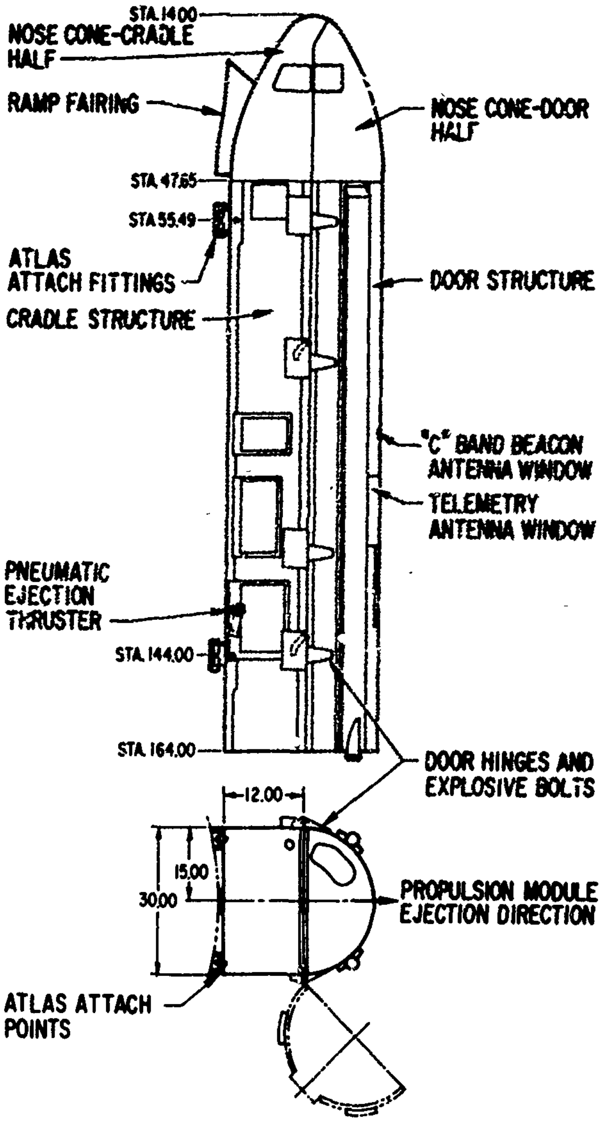


Figure 8. OV1/Propulsion Module Atlas Side-Mounting Retainer

P/M in place of the OV1 spacecraft to produce complex multiple-orbit/payload missions. For example, a single polar launch mission could involve both high elliptical (perigees >6000 n mi) and low circular (<500 n mi) orbits. Extending the concept further, one P/M on a given flight could be modified as shown in Fig. A-9 and combined with the modified OV1 spacecraft adapter of Fig. 10 to form even more complex missions. The OV1 depicted in Fig. 10 would be replaced by a non-separating payload requiring 3-axis earth orientation. The OV1 spacecraft adapter can be modified to accept up to four OV5's.

E. Centaur/Saturn Applications

The preceding concepts need not be restricted to the Atlas applications. Figure 11 shows two OV1 systems mounted on the Centaur S-V stage of the Saturn launch vehicle. The booster-retained structure of Fig. 8 is not used since the entire stage is within the Saturn fairing. Orbital buses as shown in Figs. 3 and 5 could be launched in the same fashion. Figure 12 shows possible general-utility spacecraft adaptations to the Saturn. A short

OV3/OV1 P/M orbital bus is side-mounted near the vicinity of the Instrument Unit (IU). A second OV1 system is ejected from within the IU and orients itself using the P/M GAC system prior to motor ignition. A stretched version of the OV1 spacecraft is separating from a side-mounted location and a large OV5 (15-in.) is ejecting from the interior of the IU. Prior to ejection it was stowed in the IU behind a protective door.

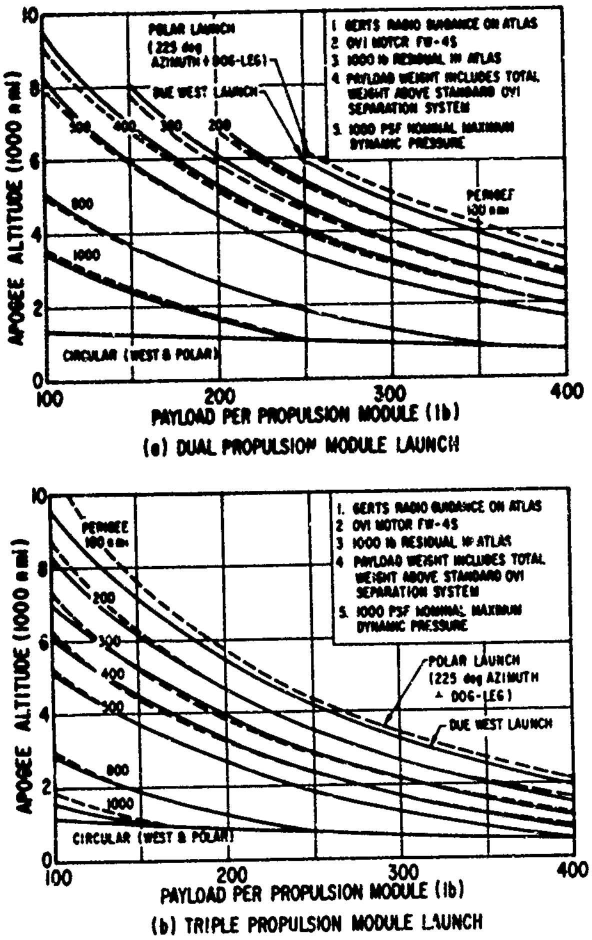


Figure 9. OV1 Propulsion Module/Atlas E/F Performance Capability

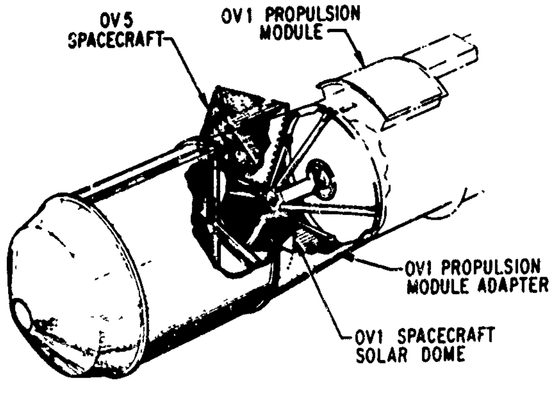


Figure 10. OV1/OV5 Multiple Payload

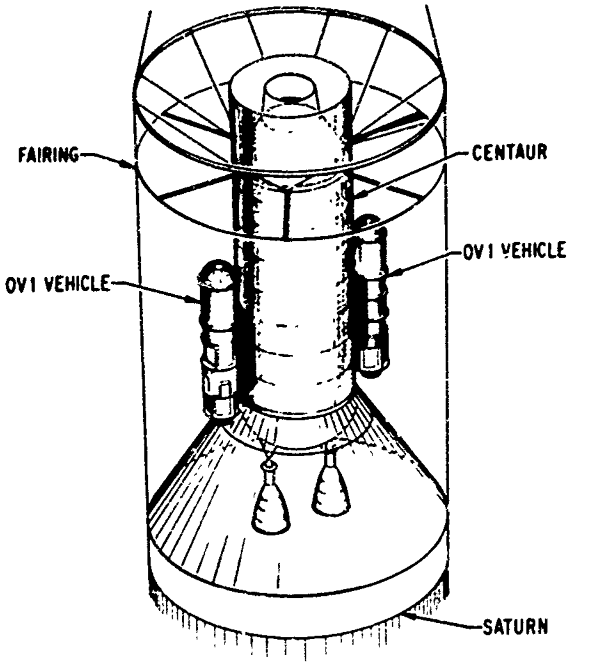


Figure 11. OV1 Vehicles Side-Mounted on Centaur

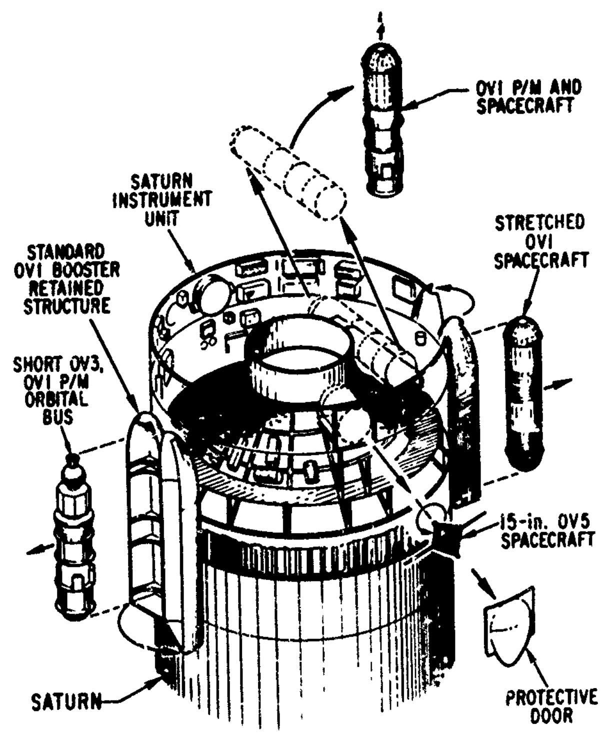
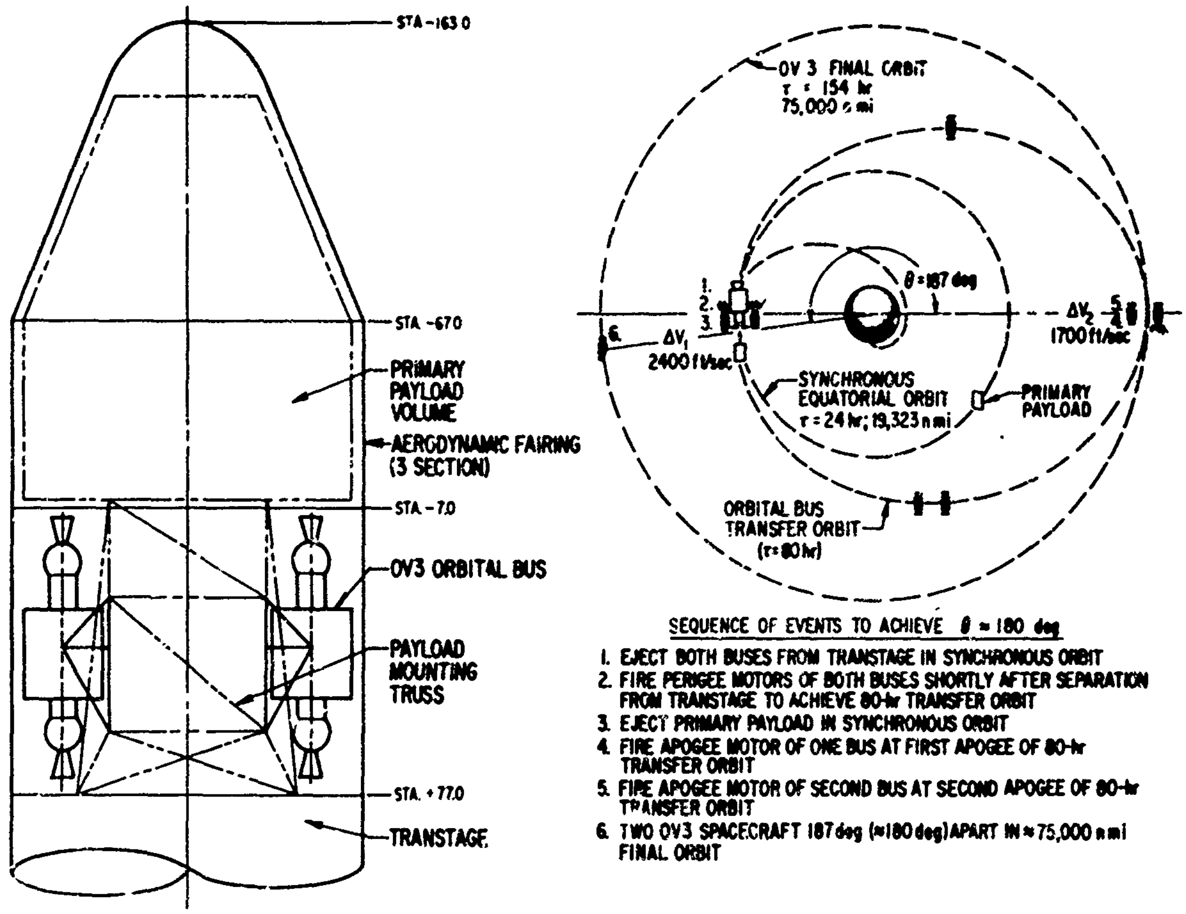


Figure 12. Saturn Multiple-Orbit/Payload

F. Titan IIIC Applications

The application of multiple-orbit/payload missions to the Titan IIIC (SLV-5C) is enhanced by the multiple restart capability of the upper stage (transtage). An interesting multiple-orbit/payload mission involving two orbital buses is illustrated in Fig. 13. The orbital buses are the same as illustrated in Fig. 3. Mission objectives are to place a primary payload in a 24-hr circular,



(a) SCHEMATIC INSTALLATION *(b) MISSION PROFILE*

Figure 13. Titan IIIC Multiple-Orbit/Payload Mission Involving Two OV3 Orbital Buses

synchronous, equatorial orbit (19,323 n mi), and two OV3 spacecraft, each carrying identical science research payloads, into a >75,000 n mi circular orbit. The two OV3 spacecraft are to be nominally positioned 180 deg (central angle) apart in their final orbit. The payloads are carried to synchronous orbit by the transtage, which initially achieves a low (≈100 n mi) parking orbit with a first burn. A second burn changes plane and injects the transtage into a Hohmann transfer orbit. At apogee of this orbit a third burn changes the orbit plane and injects the transtage into the final orbit. At this point the inertial wheels in the two orbital buses are spun up (opposite rotational directions). When both wheels achieve the desired speed, the transtage ejection sequencer is enabled. At a preselected time, the two buses simultaneously separate from the 3-axis stabilized transtage so that their longitudinal axes (thrust axis) are parallel to the inertial velocity vector. Following a short delay (≈1 min) the perigee motors of both buses are fired by ground command placing them in a Hohmann transfer orbit with an ≈75,000 n mi apogee (period ≈80 hr). At apogee of this orbit the apogee motor of one bus fires, circularizing it in the final orbit. The second bus stays in the transfer orbit and at the second apogee (1-1/2 revolutions), circularizes it in the final orbit. Due to the time delay between circularization of the two buses, the central angle separation is 187 deg, which is close enough to 180 deg to meet mission requirements. The specific sequence of events for the two buses is the same as illustrated in Fig. 4, except that all events following ejection from the transtage are controlled by ground command. This mission can be carried out at a CEI of as low as ≈ $13,000/lb.

Missions of this type can take on extreme complexity but can yield unusual cost-effectiveness when used at full potential. Overall CEI's as low as $3000/lb to $6000/lb can be achieved for configurations of the type illustrated in the final example. This mission uses the Titan IIIC transtage as an orbital launch pad, as will a short-lived spacecraft. Mission objectives are to place 27 unrelated experiments into a variety of required orbits. Of these experiments, 19 are self-contained spacecraft while 8 are experiments requiring on-orbit support (thermal control, data handling, telemetry, etc.). Nine of the spacecraft require orbits markedly different from the remainder of the experiments. An integration schematic of the experiments on the transtage and a mission profile are shown in Fig. 14.

The transtage injects directly into a 400 x 90 n mi elliptical orbit at a 38-deg inclination with a first burn. Shortly following first burn shutdown, spacecraft 1 is separated. A second burn at apogee of the initial orbit places the

transtage into a 400 n mi circular orbit with a 38-deg inclination where spacecraft 2 through 16 are separated. Spacecraft 2, 3, 4, 7, and 8 require moderate size (40 to 90-lb total weight) single-burn kick motors. Spacecraft 12 and 13 require large (>630-lb total weight) single-burn motors to achieve their final orbits; 68,000 x 400 n mi for spacecraft 12 and 400 n mi circular with a >60-deg inclination for spacecraft 13. Spacecraft 10 and 11 require orbital bus integrations to achieve their required 65,000 x 19,000 n mi elliptical orbits. The 3-axis stabilized transtage orients (or indexes) itself in inertial space to provide each payload with the required inertial orientation at the time of separation. All functions are controlled by an on-board computer and payload-eject signal sequencer.

Spacecraft 10, 11, 12, and 13 all require approximately the same size perigee kick motors. Consistent with the use of off-the-shelf hardware, Scout (or Delta) upper stages are used. This stage consists of a payload adapter (section E), an FW-4S solid propellant motor, and a spin table (section D) as a standard configuration. Spacecraft 10 and 11 require apogee motors to raise the perigee of the final orbit to the required altitude. The configuration of these vehicles is similar to that shown in Fig. 3, except that the support structure is not necessary and the lower motor is replaced with the Scout upper stage. The orbital bus thus formed mounts to the transtage payload truss with the Scout D section.

All orbital bus or kick motor payloads spin up while on the 3-axis stabilized transtage. The sequence of events for the orbital buses is identical to that shown in Fig. 4. Since spacecraft 12 and 13 do not require apogee motors, reaction wheels are not required for stabilization following perigee or kick motor separation. For simplicity, the orbital buses are controlled by ground command.

Following separation of spacecraft 16, the transtage burns a third time placing it into a 400 x 150 n mi transfer orbit. During this orbit, spacecraft 17 separates. At perigee of the orbit a fourth transtage burn circularizes the orbit at 150 n mi. Spacecraft 18 and 19 are ejected. The transtage then carries out a short-lived (≈5 day) 3-axis stabilized space test platform mission for experiments 20 through 27. Upon depletion of the attitude control system propellant, a random tumble mission (≈2 days) is carried out until battery power depletion.

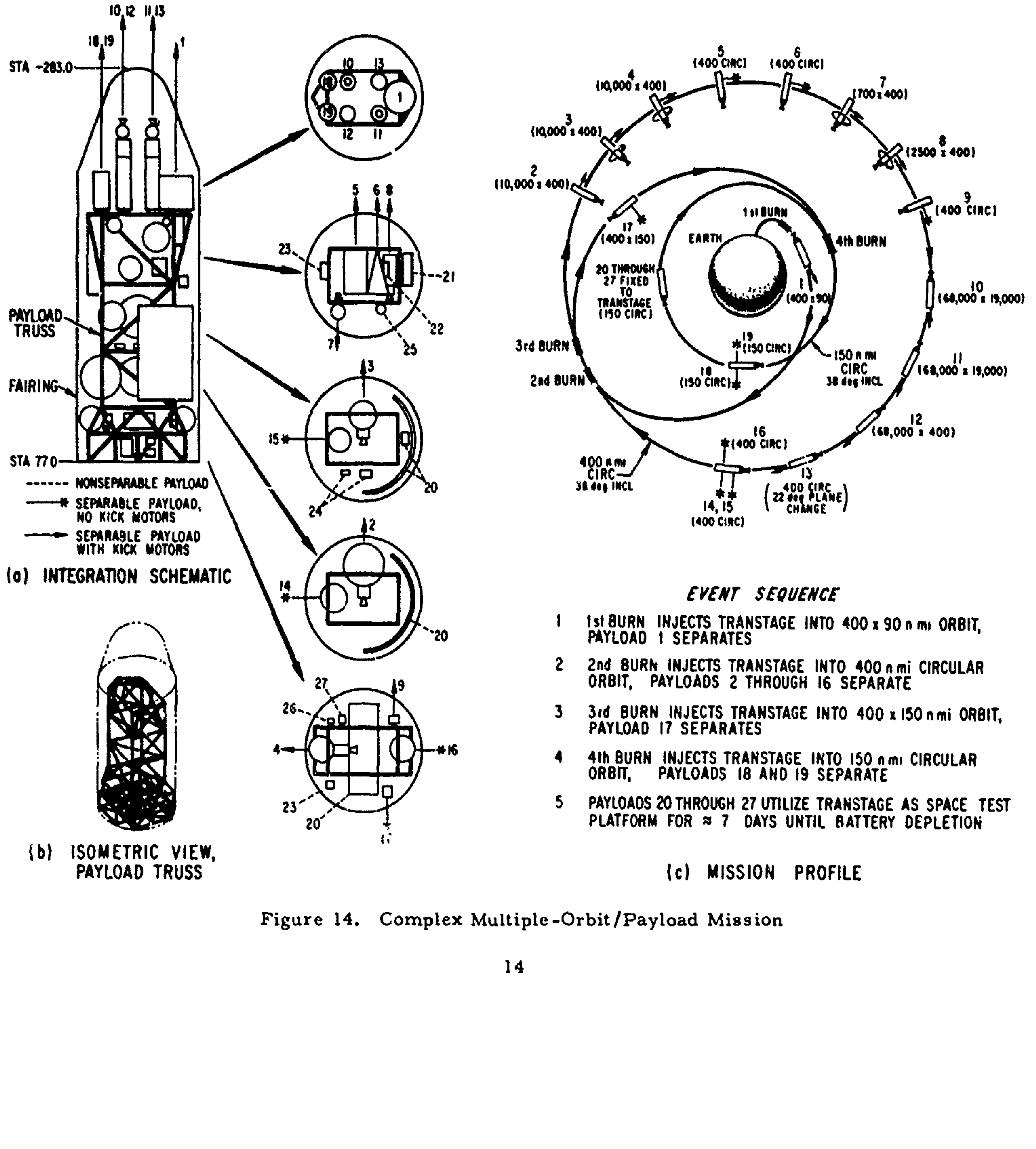


Figure 14. Complex Multiple-Orbit/Payload Mission

G. Summary

The missions defined above embody all principles of general-utility spacecraft and multiple-orbit/payload launch applications. The success of such missions hinges directly on a strong, technically competent, and creative program management and integration contractor. The technical problems are many and complex but not new. Four successful Titan IIIC multiple payload missions have been flown to date, and more will take place in the near future.

V. Conclusions

For the majority of near-earth, unmanned, space research and advanced development missions of the late 1960's and early 1970's, new spacecraft need not be developed. Mission costs can be significantly reduced by the utilization of multiple-orbit/payload launches involving general-utility spacecraft and orbital buses. These statements are based on the following facts that are supported in this paper.

1. To meet the constraints associated with flying a variety of experiments on a variety of launch vehicles in a cost-effective and timely manner, a family of general-utility spacecraft is required.

2. Previously developed spacecraft subsystems, as basic as structures, can be appropriately modified and combined with other off-the-shelf components to synthesize in "tinker toy" fashion the required general-utility spacecraft.

3. Most general-utility spacecraft can be fitted with dual or single-burn orbital buses to provide them with orbit transfer capabilities.

4. The orbital bus concept allows flexible application of the multiple-orbit/payload mission concept to small space launch vehicles, such as the Thor/Burner II, as well as large vehicles, such as the Titan IIIC.

The challenge of the 1970's is the achievement of the full potential of the multiple-orbit/payload launch concept for R&D missions. The technology and cost gains are worthy of this challenge.

VI. Acknowledgements

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APPENDIX

General-Utility Spacecraft Characteristics

This Appendix is intended to provide additional information for the evaluation of the general-utility spacecraft family concept. Brief descriptions of the current configurations and growth potentials of the OV spacecraft family are presented. The OV1 system is discussed in pages 16 - 19, the OV2 in pages 19 - 24; the OV3 pages 24 - 29 and the OV5 in pages 29 - 33. Detailed characteristics of these spacecraft are contained in the references listed at the end of this Appendix.

I. OV1 Spacecraft System

A. History

The OV1 system is an outgrowth of the OAR Atlas scientific pod program. This system, originally called SATAR, was to be carried as a secondary payload on the side of the Atlas F (SM-65). Two OV1's were launched in this manner. When the operational Atlas D (SM-65) system was phased out, a number of vehicles were assigned to the ARSP under OAR. It then became possible to mount two or three OV1's on top of a single Atlas. The combination of multiple spacecraft launch capability and the reduced cost of launch vehicles (essentially retrofit and modification costs) provided an approach that was more cost-effective than Scout launches, but devoid of the interface problems usually associated with larger launch vehicles.

A total of 10 OV1 spacecraft carrying 75 environmental sensing experiment packages have been launched for seven different agencies. Seven of these have been successful with one launch vehicle malfunction. Of these, three have completed their mission and four are still generating data. The average life to date exceeds 8 mo. The maximum demonstrated life is 18 mo. No mission failures are attributed to the spacecraft.

B. General

Two major assemblies constitute the OV1 system: (1) the spacecraft and (2) the propulsion module (P/M). Each assembly is self-contained and can be used for other applications. The P/M can be used as a small 3-axis stabilized upper stage. The spacecraft can be replaced by a multi-package dispensing unit or launched by itself aboard a number of launch vehicles. As an example of the versatility of the system, the OV1-8 P/M injected an inflatable 30-ft-diam passive sphere into orbit for radar studies, using an Atlas, whereas the spacecraft carrying a scientific payload was launched on a Titan IIIC.

C. Spacecraft

1. Configuration. The spacecraft is basically a cylinder, 27 in. in diam and 32 in. long, with faceted end domes, each 12 in. deep. The domes carry solar cells, and the cylinder constitutes the payload compartment. Primary elements of the structure are two bulkheads which form the ends of the cylinder, four longerons, and four removable panels which provide access to the payload compartment (Fig. A-1).

All electrical experiment support subsystems, except the aft bulkhead mounted battery, are located on the forward equipment shelf. Fore and aft compartments are isolated from the experiment compartment by thermal insulation panels; thus, thermal analysis of successive spacecraft is required only on the experiment compartment.

2. Weight and Volume. The OV1 spacecraft design weight is 330 lb. A typical breakdown of this weight is:

Basic spacecraft		
Structure and subsystems		110 lb
Typical payload		
Shelves and bracketry		12
Harness		8
Instruments (GFE)		200
	Total	330 lb

Payloads of up to 437 lb have been launched with only minor structural changes required. The basic spacecraft weight consists of the command telemetry and data handling equipment, electrical power system and battery, and the basic structure. The payload includes the experiment instruments, their mechanical support equipment, their electrical harness, and the stabilization system, if required.

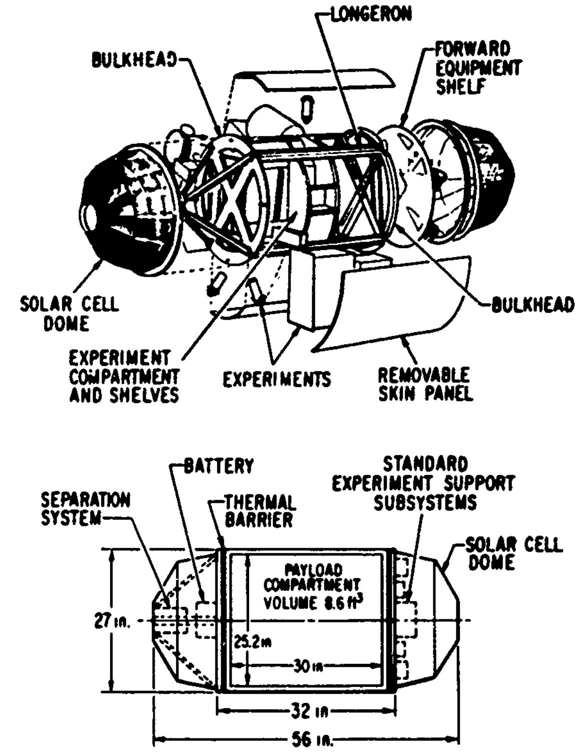


Figure A-1. OV1 Spacecraft Basic Configuration

The maximum cylindrical volume of the spacecraft is 10.2 ft^3. The interference-free volume in the experiment compartment measures 30 in. long by 25.2 in. in diam or 8.6 ft^3 as defined in Fig. A-1. This volume is typically used as follows:

Shelves, brackets, and harness	0.6 ft^3
Access volume (packaging factor: 0.76)	1.9
Instrument volume capability	6.1
Total	8.6 ft^3

3. Experiment Support Subsystems. The OV1 standardized experiment support subsystems are summarized briefly in Table A-1 and is detailed in Refs. A-1 through A-3. The subsystems utilize many off-the-shelf elements such as the transmitter, tape recorder, and command receiver/decoder from nine different manufacturers.

4. Stabilization and Orientation Subsystems. Normally, the OV1 is unstabilized and tumbles randomly. Although some OV1 spacecraft are unstabilized and tunable randomly, a 3-axis gravity gradient orientation system called a Vertistat is available for optional use to altitudes as low as 200 n mi. At these altitudes the flight attitude control system encounters significant aerodynamic drag forces. The Vertistat is designed to provide coincidence of the inherent aerodynamic reference established by a nearly circular orbit and the gravity gradient reference. The system configuration is shown in Fig. A-2. System accuracy is 5 deg in pitch and 10 deg in yaw, assuming an initial tumble rate of 1 rpm, a maximum orbit eccentricity of 2.7%, and including typical errors induced by continuous operation of on-board mechanisms and magnetic dipole moments. The system is bistable and, should initial

Table A-1. OV1 Spacecraft Payload Support Subsystems Summary

POWER
- Source: Solar array (primary) and battery
- Optimum Experiment Power: 28 W (at 25% duty cycle, 75% sunlight orbit), 100 W max
- Battery Capacity: 160 W-hr at 40% of rated capacity
- Battery Voltage: 29 to 39 V
- Voltage Regulation: 28 V ±1%
- Solar Cell Type: Silicon, blue-sensitive, N/P, 20-mil cover glass
- Battery: Ag-Cd, 27 cells, 14 A-hr at 2.8 A

THERMAL CONTROL
- Environmental Target: 0 to 120°F
- Design Approach: Passive system, thermal barriers and coatings

ENGINEERING STATUS
- Number: 5 Prime frame and 20 subframe words
- Data: Power system monitors, command verifications, temperatures, calibrations

DATA HANDLING

	PCM System*	PAM System*
• Capacity (points):	236 at 1 sample/sec 43 at 1 sample/sec	160 at 1/2×60 94 at 1/120×60
• Accuracy:	±1%	±2%
• Recorder Capacity:	240 min	120 min
• Playback Time:	15 min	7.5 min
• Clock/Time Code Generator:		
Type	Binary	Binary
Capacity	65,536 sec	163,830 sec
Resolution	1/256 sec	1/6 sec
Stability	0.01%	0.02%
Mode	Recycles to zero on record or real-time command or continuous	Continuous

TELEMETRY
- Transmitters: 1 unit, 8-W output
- Frequency: 216 to 260 MHz
- Range: 32 k bits/sec playback rate (4.8 k n mi worst case, 22.0 k n mi best case), 2 k bits/sec real-time rate (9.0 k n mi worst case, 42.0 k n mi best case)

COMMAND
- Type: IRIG
- Frequency: 406 to 549 MHz
- Antennas: Ground plane whip, near-isotropic coverage
- Number: Typically 30, 7 for spacecraft, 23 for payload

* Alternate systems

capture occur upside down, can be ground commanded to invert the spacecraft by controlled retraction of the primary booms.

D. Propulsion Module

The P/M provides structural support for the satellite, transfers flight loads to the launch vehicle structure, accelerates the spacecraft to orbital velocity or transfers orbits following separation of the system from the launch vehicle, and ejects the spacecraft after burnout of the solid fuel motor. The P/M is a complete guided 3-axis stabilized upper stage consisting of two major assemblies: (1) the electrical equipment module (motor barrel) and spacecraft adapter and (2) the attitude control module. The guidance and control equipment consists of a strapped down autopilot operating in a pulsed rebalancing mode. These elements are shown in Fig. A-3 along with the mounting of the spacecraft to the P/M.

Propulsion module performance (orbit achieved or velocity attained) is a direct function of the launch vehicle trajectory or initial orbit since the unit is a constant-impulse system. Specific orbits requiring less than maximum performance can be

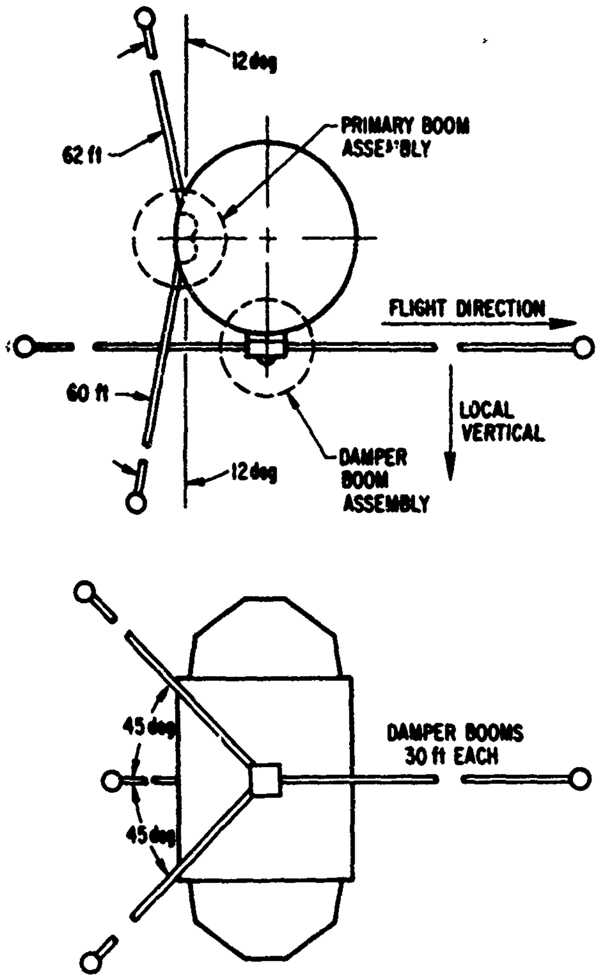


Figure A-2. OV1 Gravity Gradient System

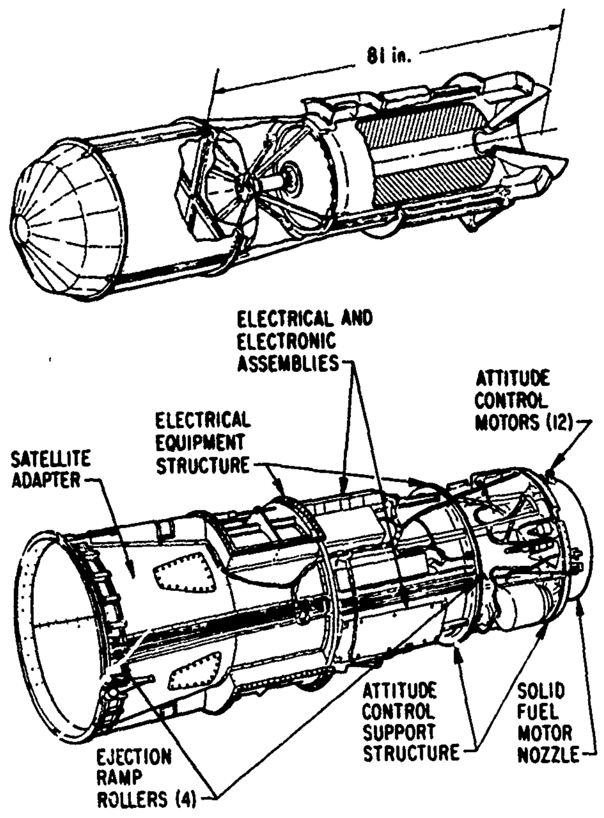


Figure A-3. OV1 Propulsion Module Configuration

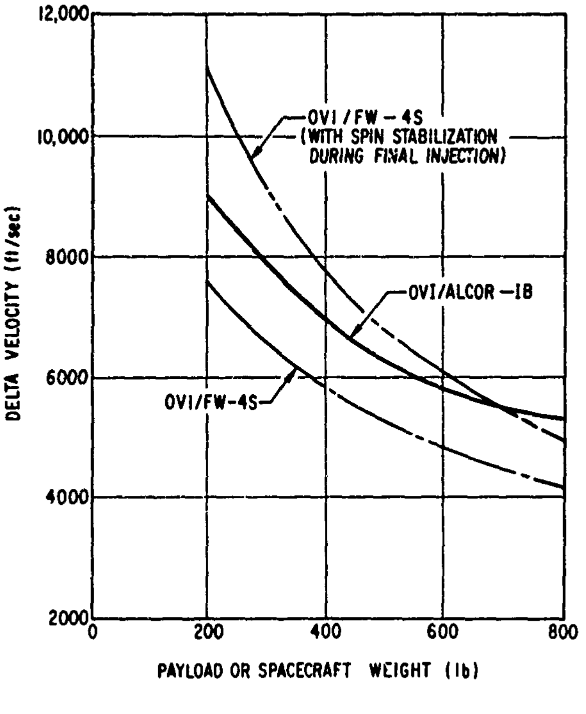


Figure A-4. OV1 Propulsion Module Performance

attained by programming to perform a plane change or pitchdown maneuver, and by using ballast. Figure A-4 defines the velocity increment as a function of payload (spacecraft) weight for the P/M with the currently used FW-4S motor and the Alcor IB motor (discussed under Growth Potential). The total weight placed in orbit is the payload weight plus the additional 278 lb of P/M structure, telemetry and tracking systems, and the empty motor case. If an OV1 spacecraft is used, the experiment payload weight is 110 lb less than the payload weight indicated. Included in the figure is the performance of the P/M with the FW-4S motor when the P/M and spacecraft are spun to 180 rpm prior to motor and payload ejection from the P/M structure and subsystems. The existing structure design is adaptable to the requirements for this separation and ejection technique. The FW-4S carries 605 lb of propellant, making a total P/M weight of 883 lb less payload.

E. Growth Potential

1. Supplementary Solar Power. The OV1 power system can be expanded by attaching 16 solar cell panels around the forward and aft bulkheads (Fig. A-5). In the stowed position, the panels would be restrained by a single wrap-around cable and released by a pyrotechnic cutter. Each panel would be extended by a torque spring and hinge. A stop-latch would hold each panel after it had moved through a 45-deg angle.

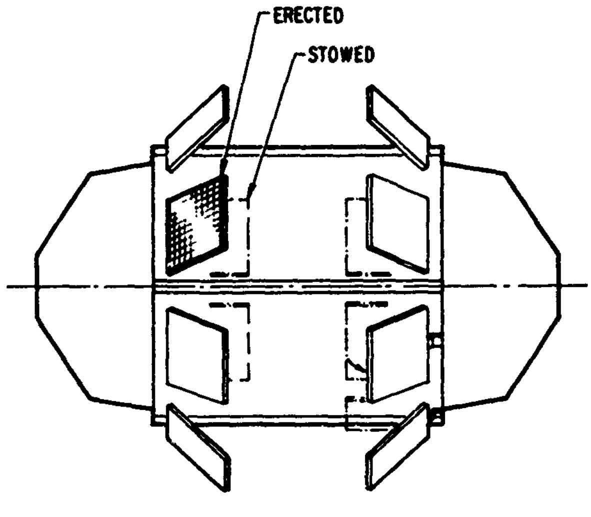


Figure A-5. OV1 Supplementary Solar Panel Installation

If unstabilized operation of the spacecraft is planned, giving essentially random orientation to the sun, the supplementary panels should provide an average 16.8-W power increase while in the sun. With the panels extended, the performance is relatively independent of the direction of the sun. For spin stabilization, particularly when the spin axis remains normal to the spacecraft sun line, the supplementary panels are more effective if hard-mounted in the stowed position. This configuration should provide a power increase of 21.4 W.

2. Spin Stabilization. The existing OV1 configuration is not satisfactory for spin stabilization since the roll axis is not the axis of maximum inertia. This problem can be dispensed with by adding three tip weighted booms (Fig. A-6). Booms 68 in. long (roll axis to weight) and weighing 4 lb can be used without undue complexity. At this radius, tip masses of approximately 9.7 lb are required to achieve stable inertia ratios. Part or all of this inert mass can be eliminated by placing experiments at the ends of the booms. Flexibility of the booms and structure or a tuned, fluid-filled, loop damper would be used to eliminate wobble following separation of the spacecraft from the P/M. Spinup would be achieved prior to spacecraft separation from the P/M by deactivating all other attitude jets and activating the roll attitude jets for a predetermined time. The P/M would be used to achieve proper inertial orientation prior to spinup.

3. Magnetic Stabilization. Magnetic stabilization similar to that employed on the Transit spacecraft can be added to the OV1 spacecraft. This system would allow the alignment of any desired axis with the local earth magnetic force field vector and reduce the angular rate about the axes normal to the magnetic vector to a low value. The system consists of: (1) a strong permanent magnet with its long axis fixed to the spacecraft structure to give the desired vehicle attitude, (2) hysteresis rods, and (3) despin shorting coils (Fig. A-7). The magnet interacts with the earth's magnetic field to provide attitude stiffness about the local magnetic vector. The hysteresis rods provide damping by interaction of angular oscillations with the vector. The despin coils consist of shorted coils of wire wound around the hysteresis rod. The low rate

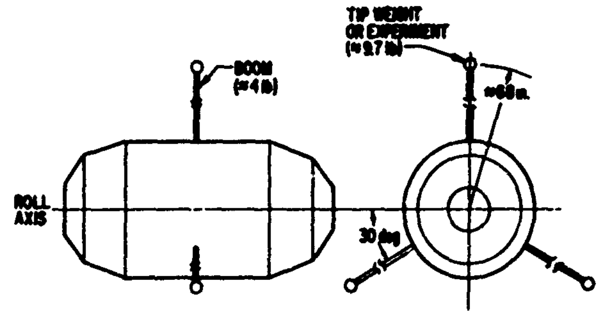


Figure A-6. OV1 Spin Stabilized Configuration

about the axis lying in a plane normal to the magnetic vector is obtained by the inherent cross-coupling of this rotation with the twice-orbit frequency rotation of the spacecraft following the magnetic vector.

Error sources and representative error magnitudes for the system of Fig. A-7 in a 350 × 200 n mi elliptical orbit are:

Source	Error (deg)
Aerodynamic and gravitational torques	0.5
Alignment with earth field	1.0
Typical on-board recorder	
Start-stop acceleration	2.0
Running	0.75
rms Total	2.4

Simulation of the system, including any error magnification due to dynamics, is required to determine the actual errors.

4. Triple Integration on Atlas Nose. The OV1 Atlas side mount structure shown in Fig. 8 (page 11) could be eliminated for 3-in-1 missions with the configuration of Fig. A-8. This modification to the system of Fig. 7 (page 11) is currently being sponsored by the OAR under the SEFSP. It is scheduled for an initial flight in late 1968.

5. Improved Propulsion Capability. The current P/M structure provides clearance for higher thrust motors such as the Alcor IB. The performance of the P/M with this motor is shown in Fig. A-4. This motor carries 911 lb of propellant, making a total P/M weight of 1189 lb less payload.

6. Orbital Platform Conversion. Studies have shown the feasibility of modifying the existing stabilization systems to convert the P/M to a 3-axis stabilized earth oriented orbital platform. These modifications are shown in Fig. A-9.

II. OV2 Spacecraft System

A. History

The OV2 spacecraft program was initiated to provide low-cost, general-utility spacecraft with a total weight range of 375 to 475 lb and a direct power output range of 70 to 120 W. The basic structural configuration and three sets of solar

panel modules were obtained from the cancelled ARPA ARENTS project.

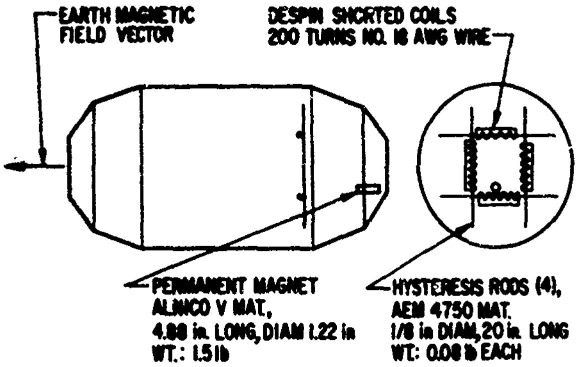


Figure A-7. OV1 Passive Magnetic Attitude Control System

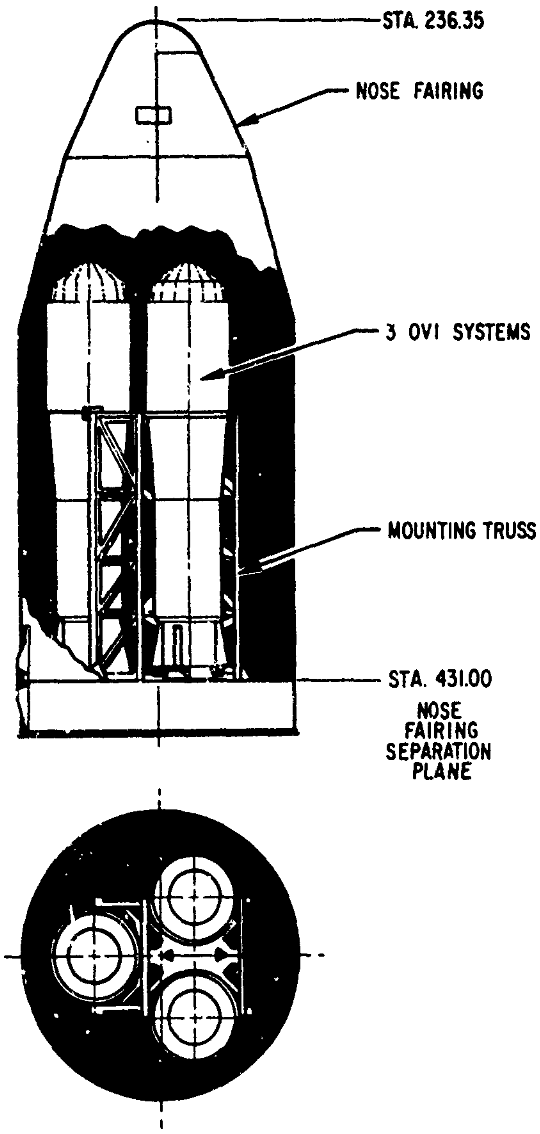


Figure A-8. Triple OV1 Atlas E/F Installation

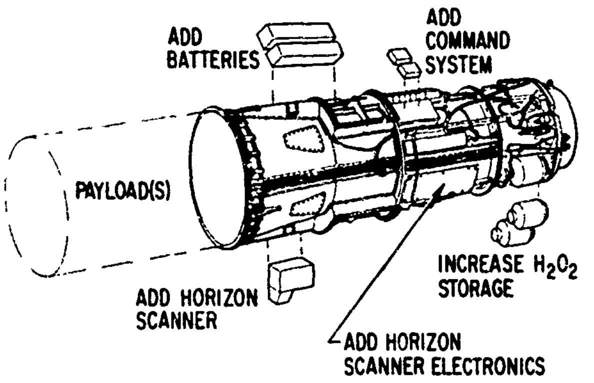


Figure A-9. OV1 Propulsion Module Orbital Platform Configuration

Two OV2 spacecraft have been fabricated and launched as secondary payloads on the R&D Titan IIIC launch vehicle. Both of these spacecraft had different orbital missions, GFE payloads, power, attitude control, telemetry and command systems. Both missions were aborted due to malfunction of the launch vehicle. The risk associated with flying payloads on R&D launch vehicles is exemplified by these failures. The third spacecraft of the program has been fabricated and is scheduled for launch aboard another Titan IIIC in early 1968. To date, 41 experiments totaling 84 hardware packages have been integrated into the three spacecraft. Each model has accommodated 12 to 14 experiments, each composed of 20 to 34 individual packages.

B. General

The OV2 spacecraft provides a near-ideal general-utility space test platform. The basic structure is a cube with an internal shelf which may be moved or entirely eliminated to accommodate component envelopes. Exterior surfaces of the structure are virtually completely available for the mounting of experiments. Solar-paddle booms allow experiment sensors to be placed at considerable distance from the structure. The basic spacecraft is magnetically "clean" (< 2 gamma at 20 ft). Both PCM and PAM data or combinations, as well as analog and digital storage equipment, are available. Telemetry equipment can include two transmitters and two receivers operating at either vhf or S-band. The power subsystem can be easily configured to match the requirements of experimental payload and support equipment. Telemetry and command are currently configured for several TT&C facilities.

The design philosophy for the program centers on the maximum use of flight-proven, off-the-shelf components. Subsystem analysis indicates a 90% overall spacecraft reliability for the first month of operation. Nominal operating lifetime is one year.

C. Configuration

The central structure of the basic OV2 consists of a 22 × 22 × 20.5-in. cube composed of six aluminum honeycomb structural panels connected by four L-section cornerposts. A seventh panel forms a center shelf within the cube. Four 30 × 38-in.

honeycomb solar paddles with cells mounted on both sides are attached to the top of the cube by hinges connected to the cornerposts. The configuration is illustrated in Fig. A-10. The basic structure has remained unchanged throughout the program to facilitate engineering analysis. The structural panels vary in thickness for different missions to optimize structural strength and weight.

During launch, the solar paddles and experiment booms are folded so the spacecraft envelope is a 64-in.-diam., 58.4-in.-high cylinder (Fig. A-11).

Separation is normally initiated by a signal from the launch vehicle. The spacecraft is also able, through an on-board g-switch-enabled timer, to separate from the launch vehicle by ground command as a backup.

When the spacecraft separates from the launch vehicle, the solar paddles and experiment booms deploy automatically. Torsional springs, shock pads, and hinge locks constitute the deployment system. Immediately following separation and deployment, the spacecraft is spun up by clusters of squib-fired rockets located on the ends of the solar-paddle spars.

D. Weight and Volume

Approximately 50 to 250 lb and 6.36 ft^3 are available for experiment packages. Four areas can be used for the location of these packages: internal and external surfaces of the cube, solar-paddle booms, and special experiment booms. Experiment location is normally determined by experiment-scan requirements, electrical or magnetic interference, heat dissipation, and mass distribution within the structure. The general characteristics of the existing OV2 configurations are summarized in Table A-2.

E. Experiment Support Subsystems

The experiment support subsystems contained in the spacecraft are the power, temperature, control, command, data handling, and telemetry subsystems. Unlike the OV1 spacecraft, these systems vary from model to model. They are summarized briefly in Table A-3 and defined in detail in Refs. A-4 through A-7.

F. Stabilization and Orientation Subsystem

The spacecraft is normally spin-stabilized, which enhances solar-paddle exposure, improves temperature control, provides an all-sky scan for experiments, and yields orientational stability. Variations of the system are presented in Table A-2. An attitude-determination subsystem is available that is capable of indicating the instantaneous orientation of the satellite to better than a 3-deg accuracy with respect to the geocenter. Precession for an OV2 is nominally a 2- to 5-deg cone half-angle. The OV2-1 was designed for an orbit with a relatively low perigee and incorporated a subliming solid-propellant system for periodic respin.

G. Growth Potential

1. Supplementary Solar Power. The power system of the basic OV2 can be easily modified by

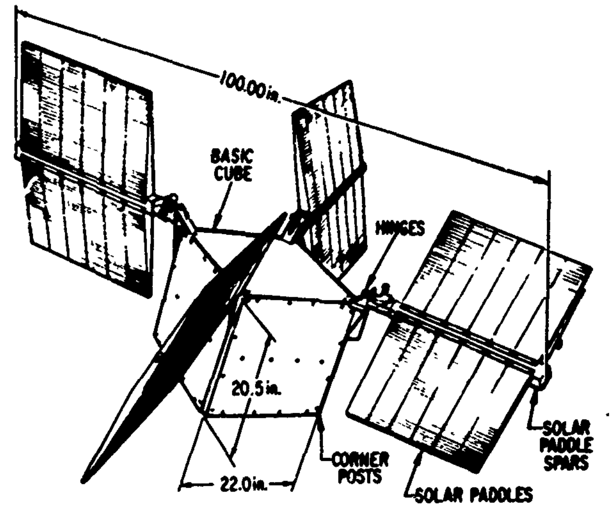


Figure A-10. OV2 Spacecraft Basic Deployed Configuration

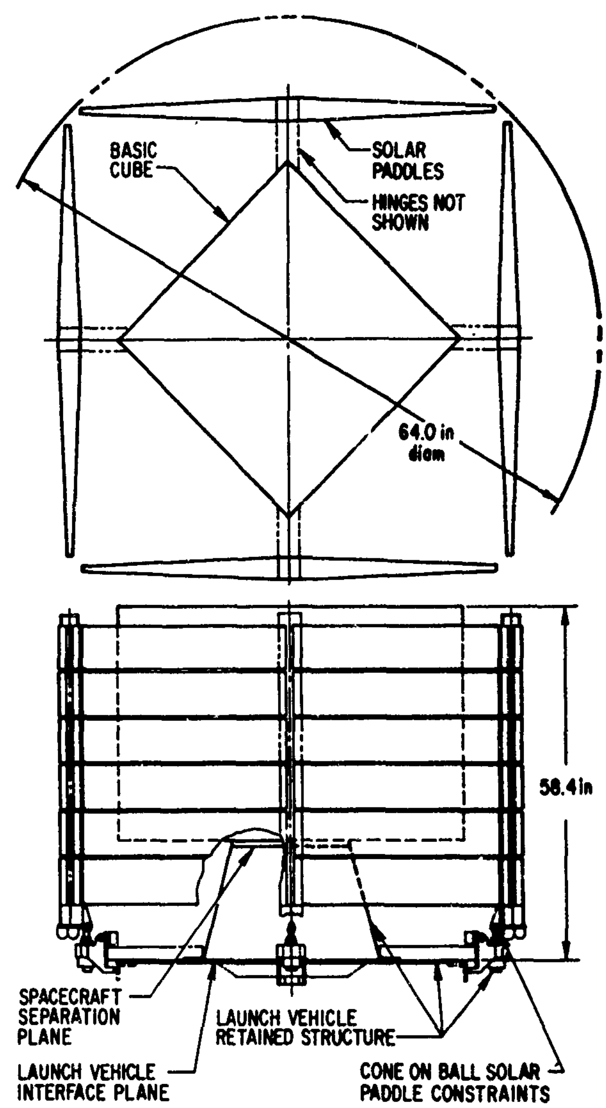
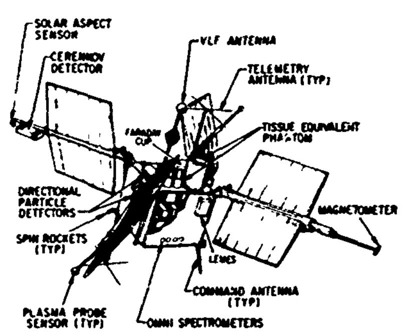


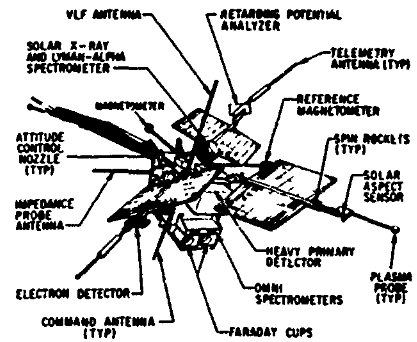
Figure A-11. Basic OV2 Folded Launch Configuration

Table A-2. OV2 Configurations

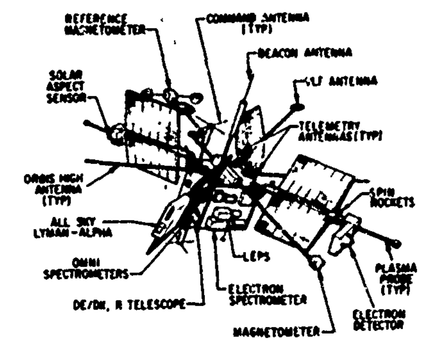
CONFIGURATION



OV2-1



OV2-3



OV2-5

	OV2-1	OV2-3	OV2-5
ORBIT	• Elliptical; 400 x 4000 n mi • 2 2-nour period, 37 deg incl	• Circular, near-synchronous; 18,200 n mi • 22-hour period; 0 deg incl, 30 deg/day drift east	• Circular, near-synchronous; 19,323 n mi • ≈24-hour period, 3 deg incl, ≈3 deg/day drift east

PAYLOAD AND SUPPORT SUBSYSTEMS VOLUME AND WEIGHT CONSTRAINTS

OV2-1

Location	Avail Volume (in.^{3})	Allow Weight (lb)	Actual Weight (lb)
Top side	7500	50	25
Upper shelf	2900	35	14
Bottom shelf	3000	45	41
Bottom side	1600	20	15
4 sides	8000	50	38
4 booms	--	25	22
Total	23,000	225	155

OV2-3

Location	Avail Volume (in.^{3})	Allow Weight (lb)	Actual Weight (lb)
Top side	7500	48	42
Upper shelf	2500	40	12
Bottom shelf	1850	20	15
Bottom side	900	15	11
4 ext sides	8000	62	52
4 booms	--	25	18
Total	20,750	210	150

OV2-5

Location	Avail Volume (in.^{3})	Allow Weight (lb)	Actual Weight (lb)
Top side	7500	48	13
Upper shelf	2500	40	5
Bottom shelf	1850	30	6
Bottom side	900	20	8
4 ext sides	8000	62	48
4 booms	--	40	38
Total	20,750	240	118

	OV2-1	OV2-3	OV2-5
TOTAL LAUNCHED WEIGHT	375 lb	425 lb	470 lb
ORIENTATION	• Inertial, non-sun oriented • Spin-stabilized at 10 rpm; respin capability • Spin axis normal to orbit plane • Precession: less than 2 deg cone half-angle • Precession damper mechanism	• Inertial, sun oriented; spin axis ±10 deg to sun line • Spin-stabilized at 4 rpm; correctional capability to maintain sun orientation • Precession: less than 4 deg cone half angle	• Inertial, non-sun oriented • Spin-stabilized at 3 rpm, no correctional capability • Spin axis in orbit plane • Precession: less than 5 deg
SEPARATION AND DEPLOYMENT	• 4 springs and single tension cable • Separation signal to cable-cutter • No free hardware • Torsional springs and locks on paddles, booms	• 3 springs and a single tension cable • Separation signal to cable-cutter • No free hardware • Torsional springs and locks on paddles, booms	• 1 spring and tension bolt • Separation signal to captive squib nut, triple redundance, no free hardware • Torsional springs, and locks and dampers on paddles, booms

Table A-3. OV2 Support Subsystems Summary

CONFIGURATION	OV2-1	OV2-3	OV2-5
POWER	• Source: Solar paddles (primary) and battery • Nominal Output: 28 V, 60 W • Paddles: Avg 74 W output; 35 W to experiments • Battery: Ni-Cd, 25% depth of discharge, 45-min discharge, 90-min charge, 2500-cycle life	• Source: Solar paddles (primary) and battery • Regulated Output: 28 ±0.3 V • Nominal Output: 28 V, 168 W total • Solar Paddle Output: 139 W (in sun orientation) • Battery: Ni-Cd, 65% depth of discharge, 6-hr discharge, 18-hr charge, 500-cycle life	• Source: Solar paddles (primary) and battery • Max Output. 28 V, 223 W (real-time mode) • Regulated Output: 28 ±0.3 V • Solar Paddle Output: 75 W average • Battery: Ni-Cd, 75% depth of discharge, 6-hr discharge, 5-hr charge, 50-cycle life
TEMPERATURE CONTROL	• Environment Target: Internal Cube 70 ±20°F; External Cube 70 ±30°F; Paddles -30 to +130°F; Battery 100°F • Design Approach: Passive system - selected coatings, shields, locations	• Environment Target: Internal Cube 70 ±30°F; External Cube 70 ±60°F; Paddles -100 to +140°F; Battery 100°F • Design Approach: Passive system - selected coatings, shields, locations	• Environment Target: Internal Cube 70 ±60°F; External Cube 70 ±60°F; Paddles -100 to +140°F; Battery 100°F • Design Approach: Passive system - selected coatings, shields, locations
ENGINEERING STATUS	• 13 Prime data points on 1/2×60 deck • 10 Secondary, subcomutated points • Status Points: Voltage (battery, system), current (battery, system); temperature (battery, solar paddle); deployment (2 booms)	• 19 Prime data points on 1/30×60 deck • 5 Secondary points • Status Points: Voltage (battery, system); current (battery, system); temperature (battery, transmitter, shelf, paddle, tank); tank pressure; valves status; boom deployments	• 14 Prime (+15 repeat) points on 1×60 PAM deck • Redline (housekeeping) data have priority output path • Status Points: Voltage (battery, system) reg and unreg; current (battery, system); temperature (battery, transmitter, shelf, paddle, decoder); boom deployments
DATA HANDLING	• All analog, PAM/FM/FM • 6 Data decks, 360 data points input • Recorder: 12:1 P/B ratio, 3-hr storage capacity	• Analog, PAM/FM/FM; Digital, PCM/FM • 3 Data decks: 226 analog, 34 digital functions • Data switch provision for transmitter failure • No recorder - real-time transmission only • Data Rate: 1440 bits/sec, 180 words/sec	• Analog, PAM/FM/FM-FM/FM-FM; Digital, PCM/FM • 3 Data decks: 224 analog, 19 digital functions • Data Readout Modes: Real-time transmit, recorder P/B, vlf clear channel, broadband hi-rate, and combinations of these • 6-hr storage capacity, 4:1 P/B ratio
TELEMETRY	• 2 FM transmitters (215 to 260 MHz) • Input power 24 W ea, output 2 W ea • Tracking beacon - normally on (400 MHz)	• 2 FM transmitters (215 to 260 MHz) • Input power 24 W ea, output 5 W ea • No tracking beacon	• 2 FM transmitters (2.2 to 2.3 GHz); one for PAM/FM, one for PCM with interchange • Input power 60 W ea, output 5 W min • Tracking beacon (136 MHz)
COMMAND PROGRAMMER	• 2 Command receivers (100 to 150 MHz) • Command Decoder: 7 real-time commands • Programmer Inputs: From decoder, recorder (cycle), transtage (separation signal); Outputs: to squibs, power subsystem, data subsystem, and timer output • Real-time verification	• 2 Command receivers (450 MHz) • Decoder: 24 real-time commands, 17 available to experiments • Programmer Inputs: From decoder, power control module (undervoltage signal), and transtage (separation signal); Outputs: to squibs, power subsystem, data subsystem, experiments, and timer output • No primary command verification	• 2 Command receivers (375 MHz) • Decoder: 32 real-time commands, 14 available to experiments • Programmer Inputs: Same as OV2-3, plus end of record and end of P/B signals from records • Data Degradation Check: Real-time + record • Individual "On" to experiments • Command verification (non-distinct)

adding to or subtracting from the solar cell modules on the paddles. The angle of the solar paddles with respect to the cube can also be changed as on the OV2-3 to obtain more power for the sun-oriented stabilization. If the spacecraft were designed to orient its spin axis perpendicular to the sun line, maximum power would be obtained by oreinting the solar paddles perpendicular to the sun line.

2. Gravity Gradient Stabilization. A 3-axis gravity gradient system similar to the OV1 Vertistat has been investigated for use with the OV2 for altitudes around 250 n mi. Use of the system would necessitate revising the angles on two solar paddles so that the top of two of the paddles on the same side of the spacecraft are facing each other (Fig. A-12). This would be necessary to balance the aerodynamic drag forces and prevent "propellering" of the spacecraft caused by the effect of these forces. The accuracy of the system is estimated to be <10 deg in all three axes.

3. Tracking, Telemetry, and Command. The tracking, telemetry, and command systems of the OV2 have been configured to meet the payload and tracking network requirements and have been different for all three models. Adaptation of the spacecraft to other networks, such as the NASA STADAN network, should not be considered a limitation of the spacecraft.

III. OV3 Spacecraft System

A. History

Since the OV3 spacecraft program was initiated in Nov 1964, four spacecraft with different payloads have been successfully orbited, demonstrating that the design meets the initial goal of producing a standardized, but versatile, system that can be easily modified to accommodate varying experiment requirements. Although initially designed for compatibility with the Scout, it is also compatible with other launch vehicles (e.g., Titan IIIC, Thor/Burner II, etc.).

B. General

The OV3 features a simple electrical and mechanical design using reliable off-the shelf hardware and considerable growth potential to accommodate a variety of experiment requirements. The standard configuration is magnetically clean enough to allow the use of magnetometers for aspect determination in low-earth orbits. The performance history of the existing four OV3's is summarized in Table A-4.

C. Configuration

The basic configuration of the OV3 (Fig. A-13) is a right octagonal cylinder 29 in. across the points and 29 in. high. The primary load-carrying structure consists of a central sheet-metal launch vehicle adapter tube, an equipment shelf of 1-in. thick aluminum honeycomb, and four load-carrying struts. The top surface of the shelf carries the payload, while the bottom surface is used for mounting the payload support subsystems. The struts stiffen the shelf, reduce the launch-induced loads to the payload, and lower the aft-solar-panel temperature by conducting heat to the side stringers. The outer shell is supported by stringers and endplate frameworks that attach to the equipment shelf at eight corners. At the top of the spacecraft is a Z-ring, supported by four tubular struts, that permits identical end plates to be used. This ring supports payloads requiring a field of view along the spin axis.

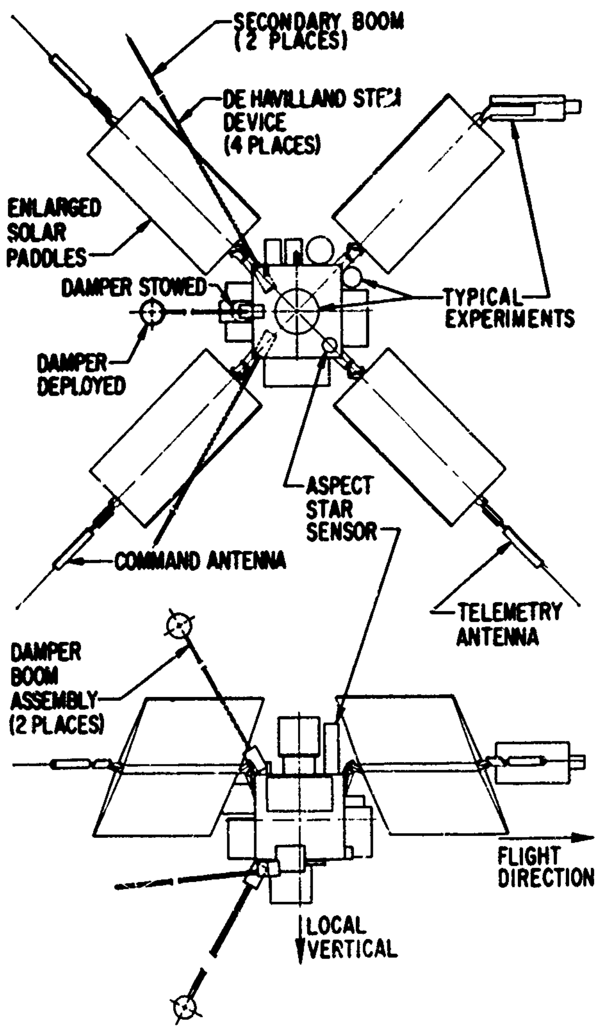


Figure A-12. OV2 Low Altitude Gravity Gradient System

Table A-4. OV3 Spacecraft Performance History

	OV3-1	OV3-2	OV3-3	OV3-4
Gross weight (lb)	151.8	200.5	165.4	171.1
Peak solar power (W)	30	43	33.5	33.5
Battery capacity (A-hr)	6	12	6	6
Launch date	22 Apr 66	28 Oct 66	4 Aug 66	10 Jun 66
Orbital data				
Altitude (n mi)	3091×195	863×172	2419×195	2554×347
Inclination (deg)	82.5	82.0	80.5	40.8
Spacecraft performance				
Experiment data	Excellent	Excellent	Excellent	Excellent
Support subsystems	Excellent	Good*	Excellent	Excellent

*Response to commands has been abnormal. As of July 1967, some pecularities are indicated, however, all experiments are operating and normal data are being retrieved

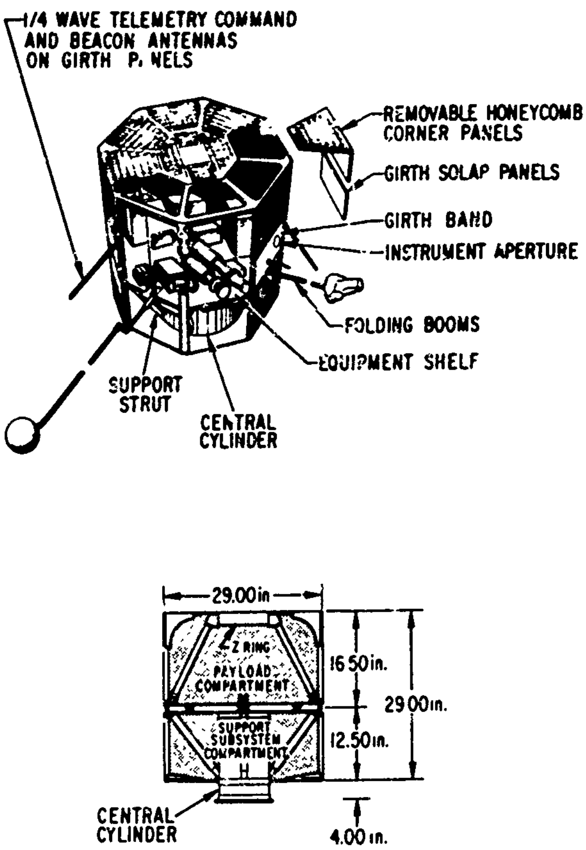


Figure A-13. OV3 Spacecraft Basic Configuration

The entire structure is covered with honeycomb panels. There are 40 individual panels with 2 basic shapes; corner and girth (Figure A-13). All panels can be individually modified as required for sensor viewports or antenna, sensor, and solar cell mounting.

The standard solar array consists of the 16 identical corner panels, 24 identical girth panels, a single top octagonal panel, and a single round panel mounted in the center of the support tube. Accessibility to the interior equipments is readily obtained by removal of the panels.

If the spacecraft is spin-stabilized, the payload must be statically and dynamically balanced. Balance is achieved in three ways: (1) appropriate location of the payload and support subsystems, (2) packaging the battery in two units located on radial lines 90 deg apart, and (3) balance weights, Where possible, the roll moment of inertia is made larger than the pitch or yaw moment for inertial stability.

Standard single- and double-fold booms are available. The single-fold booms extend a sensor 18 in. from the outer surface of the shell; the double-fold booms place a sensor 59 in. from this surface. Sensors are stowed above or below the spacecraft during launch and deploy after the separation and despin operations, if a spin-stabilized launch vehicle is used A pyrotechnically actuated latch mechanism restrains and releases the booms.

A yo-yo despin device is available for launches aboard spin-stabilized launch vehicle final stages. The assembly is located at the center of the structure and wraps around the folded booms. The yo-yo is deployed by pyrotechnic cable cutters. Further despin is effected when the booms (if used) are extended.

D. Weight and Volume

The OV3 can accommodate up to 100-lb of experiments distributed over the payload shelf. The gross spacecraft weight, less the payload, is approximately 105 lb. A typical breakdown is:

Structure	24.8 lb
Mechanical support systems	8.4
Electrical support systems	51.3
Solar array	20.5
Separable total (without payload)	105.0 lb

Approximately 5.46 ft^3 of volume and 3.64 ft^2 of area are available for the payload. This volume can be increased by allowing the payload to protrude beyond the normal external surfaces of the spacecraft.

E. Payload Support Subsystem

The standard support subsystems for the spacecraft are summarized briefly in Table A-5 and in detail in Ref. A-8. Certain modifications of the basic systems can be provided to support unusual requirements.

F. Stabilization and Orientation Subsystems

The OV3 is normally spin-stabilized. Extendable weighted booms are used to achieve favorable inertia ratios for stability if necessary. In cases where slow tumble or magnetic stabilization is required, the inertia ratio can be made to approach unity.

For spin configurations, a precession damper is used. The damper consists of a tank of mercury, an explosive valve, and a curved tube. The mercury is contained in the tank until after despin, at which time the valve actuates and allows the mercury to flow into the tube dissipating energy by friction to accomplish the required damping.

The spin rate and spin axis orientation relative to the local magnetic field vector can be derived from magnetometer data. Solar aspect sensors are available to define the orientation of the body frame relative to the sun line. Combined with the magnetometer and ephemeris data, the solar aspect data completely define the inertial orientation of the spin axes.

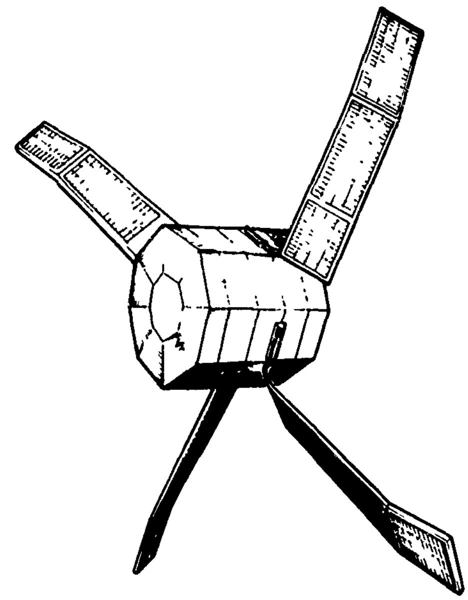
G. Growth Potential

Possible OV3 modifications, combinations of modifications, and additions are presented in Table A-6. The following discussion defines each of the modifications cited.

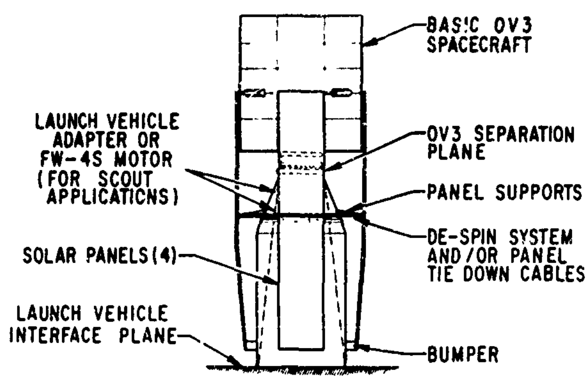
1. Supplementary Solar Power. Four paddles 10.75 in. wide by 61 or 86 in. long with solar cells on both sides can be added to the basic OV3 for increased power. The lengths are chosen for compatibility with the standard and 155-in. long Scout shrouds; however they can be any size desired.

Table A-5. OV3 Spacecraft Payload Support Subsystem Summary

POWER • Source: Solar array (primary) and battery (if required) • Battery Voltage: 23 to 30 Vdc (26.5 avg); regulation as required by payload • Output: 46 W max, 33 W avg, 20 W min • Battery: 20 6-A-hr cells, 35% depth of discharge, charge regulator • Solar Cells: N/P silicon, 20-mil quartz cover, 52 strings of 72 cells, appropriately coated solar series isolation diodes for individual solar cell strings
THERMAL CONTROL • Environmental Target: Internal equipment -10 to +90°F; external -70 to +140°F • Limitation: Payload temperature must be in the range 0 to 90°F. Active systems (thermostatically controlled blankets) can be used on local areas requiring tighter limits • Design Approach: Passive systems of selected coatings, shields, locations
ENGINEERING STATUS • Number: 13 prime points • Data: Boom deployment, structure, solar array, and battery temperatures; command receiver AGC; solar panel current; command conditioner status, experiment power monitors, battery voltage; charge and discharge currents
DATA HANDLING • Type PAM/FM/FM - FM/FM, IRIG bands 7, 11, 12 for real time data and A, C, and E for P/B • Commutators: NRZ format, one 1 X 120 and one 1 X 30, 122 points total for payload • Tape Recorder: 40% deviation FM system; 150-min record, 9.4-min P/B (16:1) • Time Code Generator: 24-hr reset, 4-sec state change
TELEMETRY • Transmitters: 1 unit, 2 W output, 17 W input • Frequency: vhf, 216 to 265 MHz • Antennas: Canted monopoles; near-isotropic coverage • Range: 3200 n mi with i. f. bandwidths of 300 kHz for real-time operation, 500 kHz for P/B operation • Tracking: CW beacon, 150 mW output, 1 W input, 216 to 265 MHz; canted monopole antenna, near-isotropic coverage
COMMAND • Type: IRIG • Frequency: 430 MHz • Antennas: Canted monopole • Number: Total 15; 7 for spacecraft operation, 8 for payload



(a) ON-ORBIT DEPLOYED CONFIGURATION



(b) LAUNCH VEHICLE STOWED CONFIGURATION

Figure A-14. OV3 Spacecraft Supplementary Solar Paddles

The characteristics of the 61- and 86-in paddles are listed in Table A-7, columns 1 and 2. The on-orbit deployed configuration is shown in Fig. A-14a.

The paddles would consist of aluminum honeycomb mounted on an aluminum longitudinal spar and covered with fiberglass face skins. The skins would be attached to a light-gauge aluminum channel section at the forward end of the paddle that would, in turn, be attached to a support rod and torsion spring system. The longitudinal spar would transmit launch forces to a load takeout bracket that distributes the load into the launch vehicle through an adapter similar to the Scout "E" section (Fig. A-14b). Paddle tie-down would be provided by a spring cable system. An explosively actuated cable cutter would sever the tie-down cable, and springs at each of the eye and clevis paddle connections would retract the tie pin releasing the paddles. A bumper would be bonded to the aft end of the paddle to eliminate flutter during launch.

A despin yo-yo similar to that used on the basic OV3 would be employed with the solar paddles if a spin stabilized launch vehicle final stage were used. The cables would be wrapped around the outside of the paddles in their stowed position. The yo-yo would be located at the plane of the load take-out bracket, and release would occur before separation of the spacecraft from the final stage of the launch vehicle. Final despin would occur during paddle deployment.

Full paddle deployment would be assured through the use of a torsion spring. A bumper stop and positive lock latch would be provided at the 90-deg rotation point. A viscous damper would be used to attenuate panel deployment shocks.

2. Short Structure. An increase in payload volume and power capacity could be achieved by placing a shorter (modified) OV3 between the payload separation plane and the basic OV3. The short OV3 could remain attached to the basic OV3 or be detached as a separate spacecraft.

The basic 29-in. long OV3 is shortened by removing two girth panels and decreasing the longitudinal stringer length of the basic structure to

Table A-6. Possible OV3 Spacecraft Modifications and Additions Summary

	Configuration	Average Solar Power Available (W)	Max Weight Available for Experiments (lb)	Max Volume Available for Experiments (ft^3)	Basic S/C Weight (Excluding Experiment) (lb)	Orientation	Stabilization
1.	Short OV3	27	60	1.79	90	Space	Spin 2-Axis
2.	Basic OV3	34	130	4.78	120	Space	Spin 2-Axis
3.	Basic OV3 and Gravity Grad Stabilization	34	112	4.78	138	Earth	2-Axis
4.	Basic OV3 and Magnetic Stabilization	34	123	4.78	127	Earth	2-Axis
5.	Basic OV3 and Attitude Control System	34	106	4.78	144	Controllable	3-Axis
6.	Basic OV3 and Aux Equip Rack	34	121	9.37	129	Space	Spin 2-Axis
7.	Basic OV3 and Short Solar Paddles	110	102	4.78	148	Space	Spin 2-Axis
8.	Basic OV3 and Short Solar Paddles and Gravity Grad Stabilization	110	84	4.78	166	Earth	2-Axis
9.	Basic OV3 and Short Solar Paddles and Magnetic Stabilization	110	95	4.78	155	Earth	2-Axis
10.	Basic OV3 and Short Solar Paddles and Attitude Control System	110	78	4.78	172	Controllable	3-Axis
11.	Basic OV3 and Short Solar Paddles and Gravity Grad and Aux Equip Rack	110	75	9.37	175	Earth	2-Axis
12.	Basic OV3 and Short Solar Paddles and Magnetic Stabilization and Aux Equip Rack	110	86	9.37	164	Earth	2-Axis
13.	Basic OV3 and Short Solar Paddles and Attitude Control and Aux Equip Rack	110	69	9.37	181	Controllable	3-Axis
14.	Basic OV3 and Long Solar Paddles	140	94	4.78	156	Space	Spin 2-Axis
15.	Basic OV3 and Long Solar Paddles and Gravity Grad Stabilization	140	76	4.78	174	Earth	2-Axis
16.	Basic OV3 and Long Solar Paddles and Magnetic Stabilization	140	87	4.78	163	Earth	2-Axis
17.	Basic OV3 and Long Solar Paddles and Attitude Control System	140	70	4.78	180	Controllable	3-Axis
18.	Basic OV3 and Long Solar Paddles and Gravity Grad and Aux Equip Rack	140	67	9.37	183	Earth	2-Axis
19.	Basic OV3 and Long Solar Paddles and Magnetic Stabilization and Aux Equip Rack	140	78	9.37	172	Earth	2-Axis
20.	Basic OV3 and Long Solar Paddles and Attitude Control and Aux Equip Rack	140	61	9.37	189	Controllable	3-Axis
21.	Basic OV3 and Long Solar Paddles and Aux Equip Rack	140	142	9.37	165	Space	Spin 2-Axis
22.	Basic OV3 and Long Solar Paddles and Short OV3	167	190	6.57	246	Space	Spin 2-Axis

18.5-in. (exclusive of end support flanges). The modified vehicle would have the same experiment mounting surface as the basic OV3 and would provide an additional 60 lb and 1.79 ft^3 volume of payload capability, as well as an additional maximum 27 W of average power (see Table A-7, column 3).

Structural support to carry the basic OV3 would be provided by extending the modified OV3 experiment platform support tube the entire length of the spacecraft rather than terminating it at the experiment platform. The tube would be 27 in. in length and 9 in. in diam. Various ways of utilizing standard and shortened versions of the OV3, and standard and extended solar paddles, are indicated in Table A-8.

3. Auxiliary Equipment Rack. To increase the payload capacity of the basic OV3, an auxiliary equipment rack (Table A-7, column 4) could be added to the top of the OV3 by means of an adapter mounting bracket, or directly to the support flange at the bottom. The length of the rack is determined by launch vehicle shroud clearances.

One configuration places the rack beneath the spacecraft (Table A-8, column 3). For a 155-in. Scout shroud, this rack would be 27 in. long. A cylindrical aluminum tube 9 in. in diam and 0.040 in. thick is sufficient to withstand launch loads. Stiffening hat sections mounted along the length of the tube would decrease deflections by increasing the tube's natural frequency, thereby reducing dynamic resonance amplification, and would provide convenient payload attachment points. The rack and fittings would weigh about 9 lb and provide an additional 3.92 ft^2 of experiment mounting surface.

4. Gravity Gradient Stabilization. The basic OV3 and its modified versions could be fitted with off-the-shelf gravity gradient systems such as the OV1 Vertistat and the General Electric (GE) dampers. The GE damper plus the related boom (de Havilland STEM type) and release mechanism could package inside the platform support tube and would be caged during launch by three spring-loaded-pin pyrotechnically released devices. The total weight of the system and release mechanism would be 18.0 lb (Table A-7, column 5).

5. Magnetic Stabilization. A magnetic stabilization system similar to that described for the OV1 spacecraft on page 19 can be added to the basic OV3. Errors are similar to those indicated for the OV1.

Table A-7. Advanced OV3 Subsystem Modules

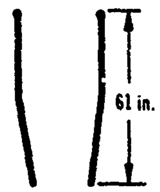
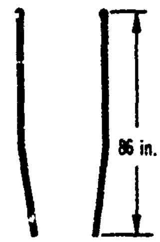
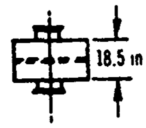
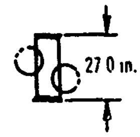
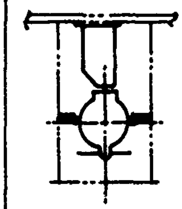
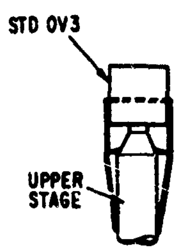
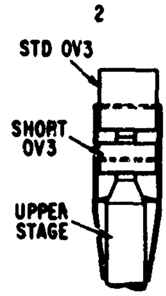
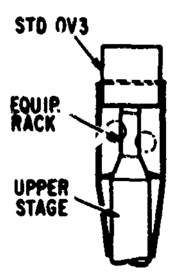
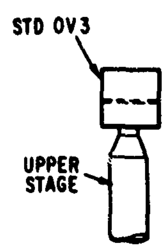
	1	2	3	4	5
MODULE CONFIGURATION					
DESCRIPTION	Standard solar cell paddle. Qty = 4	Extended solar cell paddle. Qty = 4	Short (18.5 high) OV3 satellite	Auxiliary equipment rack	G.E. gravity gradient assembly
MODULE TOTAL WEIGHT (lb)	28	36	90	9	18
MAXIMUM TOTAL POWER (W)	93	131	32	Battery powered	NA
AVERAGE TOTAL POWER (W)	76	106	27	Battery powered	NA
COMPONENTS INCLUDED IN WEIGHT TOTAL & COMMENTS	4 Solar paddles. 4 hinge and latch mechanisms. 80% solar cell coverage on paddles. 36.4 ft^2 area. Used with standard Scout shroud.	4 solar paddles. 4 hinge and latch mechanisms. 80% solar cell coverage on paddles. 51.4 ft^2 area. Used with 155 in. Scout shroud	Satellite includes standard OV3 corner panels and girth panel. Strengthened support tube capable of carrying a standard OV3. Full coverage of solar cells.	9-in. diam tubular member capable of mounting equipment on its side and supporting a standard OV3.	De Havilland extendible boom assembly with G.E. viscous damped gravity gradient assembly. Brackets, latch, and release mechanism.

Table A-8. Advanced OV3 System Configurations

	1	2	3	4
LAUNCH CONFIGURATION				
DESCRIPTION	Standard OV3 with standard paddles	Short OV3, standard OV3 with extended paddles	Auxiliary equipment rack, standard OV3 with extended paddles	Standard OV3
TOTAL SPACECRAFT WEIGHT (lb)	148	246	129	120
MAXIMUM TOTAL POWER (W)	136	Standard OV3 = 174 Short OV3 = 28	174 (solar cells only)	43
AVERAGE TOTAL POWER (W)	110	Standard OV3 = 140 Short OV3 = 27	140 (solar cells only)	34
COMPONENTS INCLUDED IN WEIGHT TOTAL & COMMENTS	Standard OV3 structure. 4 standard solar cell paddles. 1 gravity gradient assembly. Maximum no. of solar cells, body and paddles.	Standard OV3 structure. 4 extended solar cell paddles. Short OV3. 2 gravity gradient assemblies. Maximum no. of solar cells, body and paddles. Despin mechanism.	Standard OV3 structure. Auxiliary equipment rack structure. Extended solar cell paddles. 1 gravity gradient assembly. Maximum no. of solar cells, body and paddles. Despin mechanism.	Standard OV3 structure. 1 gravity gradient assembly. Maximum no. of solar cells on body. Despin mechanism.

6. Active Attitude Control. Active attitude control systems currently used on spacecraft include reaction wheels and gas jets. In the category of reaction wheels, several types can be used for angular momentum storage, e.g., inertial wheels and fluid flywheels. A comparison of the subsystem elements for these systems is presented in Table A-9, along with a comparison of the elements integrated into the OV3 as 3-axis attitude control systems. The inertial wheel subsystem weighs 8 lb less than the fluid flywheel subsystem. However, the fluid flywheel subsystem has distinct power and reliability advantages and is recommended for the OV3 system.

The fluid flywheel consists basically of a dc conduction pump which pumps mercury through a closed loop of stainless steel tubing producing a control torque on the spacecraft as long as there is a rate of change of speed for the mercury flow. A power converter is required for the pump. There are no bearings in the system. If an electromagnetic pump is employed instead of the conduction pump, there would be no moving mechanical parts. The tubing is routed about the structure of the vehicle, within bend radius limits, leaving the center of the vehicle unobstructed.

7. PCM Data Handling. The OV3 can be readily converted to a PCM data handling system such as those used on the OV1 and OV2 spacecraft. An interesting high data rate, low error, low power system termed Digilock, is described in Ref. A-8.

IV. OV5 Spacecraft

A. History

The OV5 series spacecraft are part of a larger family of proven general-utility "minispacecraft" called Environmental Research Satellites (ERS) intended to be orbited as "piggyback" or secondary payloads.

The ERS are customized for one or two experiments in order to minimize integration time and compromises of the experimental goals. The ERS concept evolved primarily because of the difficulties in obtaining flights on larger spacecraft which offer a wealth of on-orbit support necessary for many experiments, but which have attendant integration problems, long lead times, and are relatively expensive. Recognizing that some experiments can be conducted with less complex spacecraft, efforts were initiated in late 1960 to develop a completely independent system which was simple, flexible, and would impose no significant burden on any launch vehicle or primary spacecraft system. The initial program was directed toward a 1.5-lb minispacecraft carrying solar cells for radiation-damage measurements. Design and fabrication lead time was four months. The success of the program led to follow-on efforts utilizing essentially the same subsystems for different experiments and gradually to a minispacecraft family with versatile subsystem capabilities.

A total of 12 ERS sponsored by four separate Air Force agencies have been orbited as piggyback payloads (Table A-10). All have carried out their missions as designed. Four additional ORS-III ERS are currently being fabricated: two for NASA and two for the Air Force (OV5-2 and OV5-4).

Table A-9. OV3 Reaction Wheel Systems Comparison

PARAMETERS	FLUID FLYWHEEL	INERTIA WHEEL
Subsystem Elements		
• Stall torque (lb-ft)	0.2	0.01
• Saturation momentum (lb-ft-sec)	0.25	0.25
• Power at saturation torque (W)	10.0	48.0
• Average power (W)	2.0	5.0
• Subsystem weight (lb)	22.0	14.0
• Probability of success for 1 yr	0.9681	0.6537
Integrated 3-Axis Systems (Dual Modes)		
• Peak power (W)	44.0	7.0
• Average power (W)	4.0	7.0
• Weight (lb)	36.0	28.0
• Probability of success for 1 yr	0.9201	0.6213

Table A-10. ERS Flight History

ERS DESIGNATION	NO. ORBITED	SHAPE	APPROX WEIGHT (lb)	LAUNCH VEHICLE
TRS-I	4	Tetrahedron 5.5 in. on side	1.5	Thor-Agena
TRS-II	2	Tetrahedron 9.0 in. on side	4.0	Thor-Agena
ORS-II	3	Octahedron 9.0 in. on side	16.0	Atlas-Agena
ORS-III	3	Octahedron 11.0 in. on side	17.0	Titan IIIC

These will be flown on Thor/Delta and Titan IIIC launches in late 1967 and early 1968.

B. General

The ERS octahedron and tetrahedron shapes were originally selected because they provide minimal variations in projected areas, regardless of orientation relative to the sun, and thus yield nearly constant output from the body-mounted solar array. These shapes, particularly the octahedron, have also proven desirable from such viewpoints as dynamics, fabrication, testing accessibility, and stowage on the launch vehicle.

A versatile and reliable series of subsystems has been developed, which includes power (solar array and supplementary battery), telemetry, command, antenna, stabilization (0-g random tumble and passive magnetic spin), and aspect sensing. Other specialized subsystems have been provided for previous missions and can be supplied as required.

C. Configuration

The basic structure of all ERS is an aluminum framework which provides the mounting supports for both external and internal components. The octahedron frame consists of 12 formed members constituting the corner edges of the polyhedron. The tetrahedron requires six such members. Shear webs support the exterior framework and provide component mounting platforms. All members are integrally brazed to form a rigid unitized structure; however, on the octahedrons one edge member is made removable to provide access during installation of larger components. A fitting at one apex serves as a support during launch and as a guide during separation from the launch vehicle. Triangular solar cell panels are mechanically fastened to the framework and are easily removed for access to electronic circuitry. The octahedrons require eight solar cell panels and the tetrahedrons, four. The antennas consist of ordinary off-the-shelf carpenter tape elements cut to the appropriate length. Figure A-15 shows an ORS-III with four solar panels removed to reveal the structure and typical experiment integration.

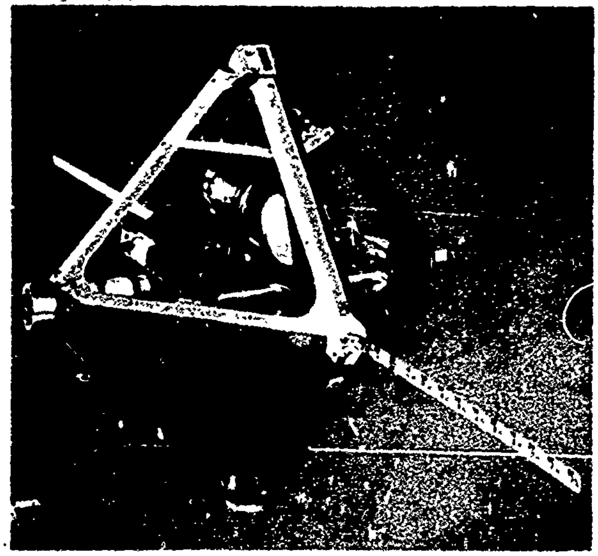


Figure A-15. ORS-III with Solar Panels Removed Showing the Structure

At launch, the ERS are mounted in a containment canister which provides support and also incorporates the ejection mechanism. The canister has a center support post and additional load support points. For the octahedron, the load support points are located at the plane of the four apexes. A pyrotechnically actuated pin puller retains the spacecraft in the canister, and on firing, initiates separation. The spacecraft is ejected from the canister by a spring at a velocity of 2 to 8 ft/sec. The antennas are stowed during launch and maintained in position by retainers on the canister. As the spacecraft ejects, the antennas automatically deploy.

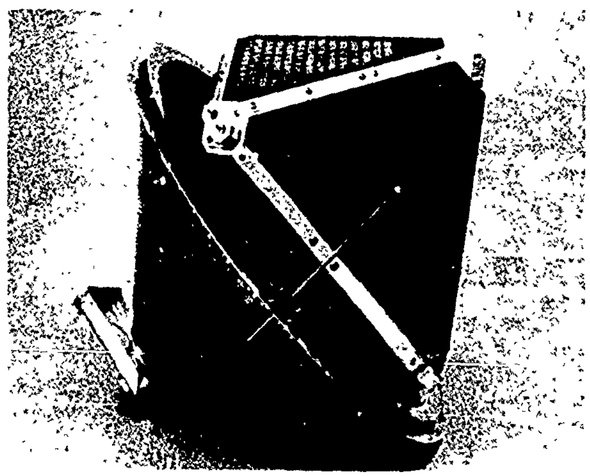


Figure A-16. ORS-III with Containment Canister

The canister is usually the only interface hardware with the launch vehicle. The interface consists of only four machine bolt fasteners and a single 22-V electrical connection for the pyrotechnic pin puller. In some cases, a microswitch is incorporated in the system to indicate spacecraft separation through the launch vehicle telemetry. Figure A-16 illustrates a typical ORS mounted in the canister. The pyrotechnics and pin puller (not shown) are mounted in the support bracket at the apex of the canister cone. The launch envelopes, including the ERS, separation mechanism, and stowage canister, for the various configurations are

TRS-I (6 in.)	7 × 7 × 7 in.
TRS-II (9 in.)	10.5 × 10.5 × 10.5 in.
ORS-II (9 in.)	11.06 × 12.38 × 10.50 in.
ORS-III (11 in.)	13.75 × 15.38 × 12.69 in.

The deployed configuration of the ERS is the basic polyhedron shape with the antennas extended to a straight dipole position. The antennas measure ≈41 in. from tip to tip.

D. Weight and Volume

The gross total weight of the ERS ranges from 1.5 to as high as 75 lb. The total weight, less payload, ranges from 1.1 to ≈15.5 lb. A weight breakdown is presented in Table A-11.

Mounting locations of the payload within an ERS are generally not critical and are determined to suit the particular payload. Some payloads have extended through and protruded from the opposite apexes of the structure. However, usually the payload is located in the central portion of the structure with the support subsystems filling in the corners. Figure A-15 illustrates a typical packaging configuration. The general characteristics of the current ERS family are summarized in Table A-12.

E. Experiment Support Subsystems

The experiment support subsystems contained in the spacecraft are summarized briefly in Table A-13 and in detail in Refs. A-9 and A-10. Figure A-17 shows allowable duty cycles for the ERS family for various payload power levels. Included in this figure is the information for a 15-in. ORS which will be discussed under Growth Potential.

F. Stabilization and Orientation System

In most cases, a torsional spring system in the ejection mechanism is used to impart a spin

Table A-11. ERS Weight Breakdown (lb)

SUBSYSTEM	TRS-I (6-in)	TRS-II (9 in.)	ORS-II (9 in.)	ORS-III (11 in.)
Payload (max)	3.0	10.0	20.0	40.0
Structure	0.6	1.7	3.2	3.8
Electrical Power Systems				
Solar panels	0.5	1.1	2.2	2.5
Battery	0.8 to 3.4			
Voltage regulator	0.2 to 0.6			
Battery charger	0.2 to 0.7			
VHF Telemetry				
Transmitter	0.2 to 0.4			
SCO	0.1 to 0.2			
Commutator	0.1 to 0.2			
VHF Command-Receiver				
Receiver	0.5			
Decoder	0.6			
Logic unit	0.3 to 1.0			
Diplexer, VHF	0.3			
Stabilization System				
Magnet	0.1			
Magnetic damper	0.1			

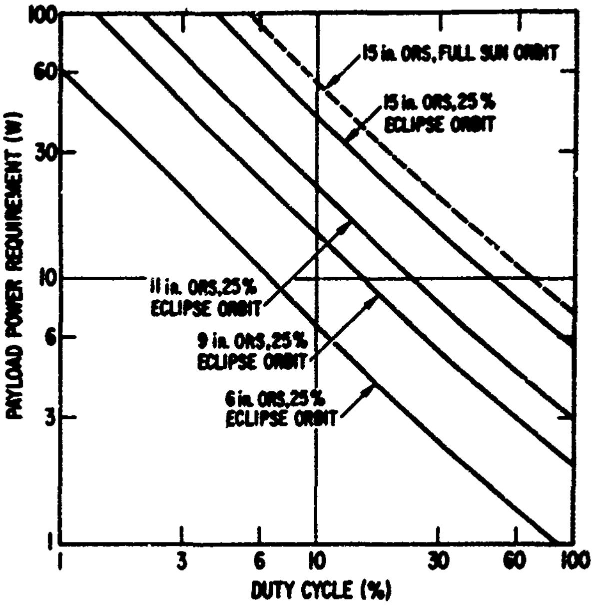
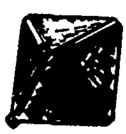
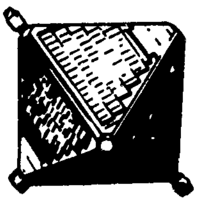


Figure A-17. ERS Allowable Duty Cycles for Battery Supplemented System

Table A-12. ERS General Characteristics

CONFIGURATION				
CHARACTERISTICS	TRS I	TRS II	ORS II	ORS III
WEIGHT	1.5 lb*	4.0 to 5.0 lb*	5.0 to 9.0 lb*	7.0 to 25.0 lb*
SIDE LENGTH	6 in.	9 in.	9 in.	11 in.
TOTAL VOLUME	26 in.^{3}	88 in.^{3}	346 in.^{3}	926 in.^{3}
PAYLOAD VOLUME	6 in.^{3}	60 in.^{3}	320 in.^{3}	400 in.^{3}
NUMBER OF EXPERIMENTS	up to 5 -- time sequenced	up to 7 -- time sequenced	up to 14 -- time sequenced	up to 14 -- time sequenced
TELEMETRY	8 chan analog -- 100 mW radiated power	8 chan analog -- 100 mW radiated power	16 chan analog -- 100 mW radiated power	32 chan analog -- 1 W radiated power
POWER	0.8 W, regulated	1.6 W, regulated	3.2 W, regulated	5 W, regulated**

*Total weight is dependent on payload weight, telemetry requirements, etc. Telemetry is designed for individual experiments.

**Command receiver and rechargable battery supply available.

(3 to 100 rpm as required) to the ERS to facilitate on-orbit thermal control, communications, and to improve solar array performance. The spring system can be removed to eliminate the spin, yielding a slow random tumble. With no intentional spin, an extremely low acceleration environment on the order of 10^{-4} to 10^{-5} g's can be obtained. Even lower accelerations can be achieved by the addition of a magnetic damping matrix defined below.

G. Growth Potential

1. Structure and Shape. ERS can be fabricated in sizes and shapes other than those currently available. The selection of a particular shape is dependent upon mission requirements. A 15-in. ORS and an 18-in. prismatic shape have undergone considerable design efforts.

Table A-13. ERS Support Subsystem Summary

POWER •Source: Solar array (primary) and battery (if required) •12 to 18 V from solar array •+4 V ± 0.1% regulator nominally used (+3, +6 available), high voltage supply available •TRS-I, 1.1 W; TRS-II, 1.9 W; ORS-II, 3.5 W, ORS-III, 5.3 W continuous from solar array •Battery: 48 W-hr capacity - 10 cell Ni-Cd permits short duration high power level and eclipse operation •Solar Cells: N/P silicon, quartz covers as required •Duty Cycle: See Fig. 15 •Undervoltage Control: Prevents battery damage
TEMPERATURE CONTROL •Design Approach: Passive, utilizing control of absorptivity and emissivity of surfaces with selected thermal control materials; heaters can be provided
ENGINEERING STATUS •Data: Temperatures (as required), unregulated and regulated voltages, currents
DATA HANDLING •Type: PAM/FM/PM or PCM/FM/PM, IRIG bands 5 and 3, other bands can be used •Commutator: Up to 32 points of 1 to 10 sec duration; format to meet mission requirements •Bandwidth: 20 Hz, higher bandwidths readily implemented •Storage: Available as required (core memories, magnetic latch relay matrices); not used as yet
TELEMETRY •Transmitter: 100 mW or 1 W radiated output, 200 mW and 2 W inputs, respectively, 1/spacecraft •Frequency: 136 to 137 MHz compatible with NASA STADAN; compatibility with USAF, NRD, and STC available on request •Antennas: Single half-wave dipole located on opposite apex of spacecraft; typical dipole patterns •Range: 100 mW - 20,000 n mi; 1 W - 65,000 n mi •Tracking: Uses telemetry •Data Accruacy: 1%
COMMAND •Type: NASA/STADAN standard; compatibility with other ranges available - fixed-tuned AM •Range: 75 k n mi •Frequency: 148 MHz •Antennas: Dipole and monopole; normal dipole coverage •Number: Up to 21; normal operation of spacecraft exclusive of experiments does not require commands; no command verification

2. Data Storage. Appropriately sized tape recorders, core memories, or magnetic latch relay matrices are commercially available and can be readily integrated into the ERS family.

3. Stabilization Systems. A number of types of stabilization systems other than random and spin are readily adaptable to the ERS family, such as: (a) passive magnetic, (b) active magnetic with provisions for torquing, (c) gravity gradient, and (d) spin vector precession. These systems can be used in conjunction with the aspect system to provide positive orientation data. A discussion of the capabilities of the three systems is given below.

a. Passive Magnetic System. This system is similar to that described for the OV1 on page 19 except the despin coils are not required. Approximate system weights are 0.11 lb for the permanent magnet and 0.09 lb for the 16 permeable rods, or a total of 0.20 lb. The roll axis may be expected to capture (±10 deg) within 6 to 20 hr.

The system is suited for aligning antennas for optimum linkage with ground stations and for pointing devices such as trapped radiation or IR detectors. For example, by placing the spacecraft into an equatorial orbit, an IR detector can be aimed with a 10-deg accuracy into the northern or southern galactic sphere; by placing the spacecraft into a polar orbit, the detector can be made to scan 360 deg twice each orbit. The scan direction would be along the direction of the earth's magnetic field vector. In this case, the spacecraft inverts each time it crosses the earth's poles.

b. Active Magnetic Systems. Active magnetic stabilization employs a spinning spacecraft with electromagnet torquing capabilities. The spinning vehicle remains oriented in inertial space unless the electromagnet is activated via a command from a ground station at which time the vehicle is torqued, or precessed, in a direction dependent on the direction of the local earth magnetic field vector. Selection of the the time for ground command is based upon ephemeris and aspect data. Subsequent corrections are made until the vehicle is "jockeyed" into the desired inertial orientation. Relatively accurate pointing can be made to almost any place in the universe, such as aiming a sensor at a sector within the Milky Way during an IR astronomy mission. The system is versatile and accurate.

c. Gravity Gradient System. This system is capable of maintaining one axis of the spacecraft pointing towards or away from the earth at all times. Pointing accuracies of 1 to 10 deg can be obtained, depending on the degree of refinement in the system.

Two basic systems have been designed but not fabricated. For low altitude missions where alignment accuracy of only 10 deg to the local vertical is required, a simple system is available which utilizes one rigid boom approximately 50 ft long and a damper consisting of permeable magnetic rods located in the structure. For alignment accuracy of 1 deg at low altitudes and for use at synchronous orbit altitudes, a quartz-fiber hysteresis-damper system would be utilized with a multi-boom array.

Examples of uses of this system are: (1) for an IR astronomy mission to permit the scanning of gradually changing discs in space for IR energy; (2) for a communication mission to permit use of a higher-gain antenna, since one axis always points towards the earth's surface. Other applications might be observational missions, such as weather, video, or uv albedo.

d. Spin Vector Precession System. This system is similar to that employed on the Vela and OV2-3 spacecraft. It is capable of aligning the spin axis of the ERS either perpendicular or parallel to the spacecraft sun line to an accuracy of 10 deg. Components have been sized for the 11-in. ORS and can be configured for other ERS. The 11-in. ORS system provides, at a 25-rpm spacecraft spin rate, one initial 90-deg orientation maneuver and 46 15-deg correction maneuvers at two-week intervals, yielding two years of orientation capability. At 10 rpm, the same system provides 130 15-deg correction maneuvers, the equivalent of five years of orientation capability.

The system consists of three primary elements: (1) a dry nitrogen storage tank, (2) a single nozzle, regulator valve and supply line, and (3) a sun sensor and electronics. For the 11-in. ORS, the 3000-psi nitrogen tank is 8 in. long and 2 in. in diameter with a volume of 25 in.^{3}. The weight of the system is:

Thrust nozzle	0.1 lb
Nitrogen tank	2.0
Nitrogen	0.3
Valve and feed lines	0.6
Regulator	0.4
Sun sensor	0.2
Electronics	0.1
Total	3.7 lb

The regulator valve is activated by ground command.

The system is suitable for a variety of missions, which include solar x-ray and cosmic ray detection, thermal coating tests, and solar cell degradation tests.

4. Eject Initiation System. Irrespective of the simplified launch vehicle interface of the ERS, instances occur where piggyback rides cannot be obtained because of the lead time and cost associated with implementing the electrical interface. This difficulty can be eliminated by providing an independent separation signal from an ejection initiate module mounted on the containment canister. This module would be entirely self-contained and attach to the containment canister in the same position as the current pyrotechnic pin puller. The module would be annular in shape and contain a battery for firing the pin puller squibs, a timer, and dual g-switches. The timer would be a solid state magnetic logic unit which would not reset as a result of rfi transients or power dropouts and would be capable of being programmed while on the launch vehicle. The battery would be a sealed primary Ag-Zn unit. The g-switches would be set to close at launch vehicle liftoff, thereby applying power to the timer which would eject the ERS at a predetermined time from liftoff. The module would weigh approximately 5 lb and be qualified to a variety of launch vehicle environments. The advantage would be the elimination of all electrical interface with the launch vehicle, allowing the spacecraft to be integrated with an extremely short lead time.

V. Appendix References

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13. ABSTRACT

Costs for the majority of near-earth unmanned, space research and advanced development missions of the late 1960's and early 1970's can be significantly reduced by using multiple-orbit/payload launches involving general-utility spacecraft and orbital buses. This concept has evolved through the implementation of the new DOD Space Experiments and Flight Support Program (SEFSP). The modification and combination of previously developed spacecraft with other off-the-shelf space flight proven hardware to synthesize in "tinker toy" fashion a general-utility spacecraft family for use in R&D programs of this nature is discussed. The current characteristics and growth potential of the low cost, general-utility OV spacecraft family (OV1, 2, 3, and 5) which utilize off-the-shelf hardware to a maximum extent are described. The concept of the orbital bus is developed. A typical R&D program involving four spacecraft, each from a different agency, is used to show that total overall program cost can be reduced by as much as 55% through the use of multi-agency, multiple-orbit/payload, single launch vehicle missions involving orbital buses. Hypothetical, typical multiple-orbit/payload missions on both large and small launch vehicles are described.

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14. KEY WORDS

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Orbital Bus
Orbiting Vehicle 1 (OV1)
Orbiting Vehicle 2 (OV2)
Orbiting Vehicle 3 (OV3)
Orbiting Vehicle 5 (OV5)

Abstract (Continued)